NASA CR71500

MISSILE AND SPACE DIVISION

VOYAGER SPACECRAFT SYSTEM

PHASE IA TASK B PRELIMINARY DESIGN

PROPULSION SYSTEM ANALYSIS

VOLUME C

VOYAGER

GENERAL ELECTRIC

GENERAL ELECTRIC

NASA CR71500

DIN 65SD4514
31 January 1966

VOYAGER SPACECRAFT SYSTEM

PHASE IA TASK B PRELIMINARY DESIGN

PROPULSION SYSTEM ANALYSIS

VOLUME C

PREPARED UNDER CONTRACT 951112
under NAS 7-100

CALIFORNIA INSTITUTE OF TECHNOLOGY

JET PROPULSION LABORATORY

4800 OAK GROVE DRIVE

PASADENA, CALIFORNIA

GENERAL ELECTRIC
MISSILE AND SPACE DIVISION
Valley Forge Space Technology Center
P.O. Box 8555 • Philadelphia 1, Penna.

TABLE OF CONTENTS

SUMMARY OF PHASE IA TASK B REPORT

VOLUME A - SPACECRAFT FUNCTIONAL DESCRIPTION (2 Books)

Section	
I	Mission Objectives and Design Criteria
II	Design Characteristics and Restraints
III	System Functional Descriptions
IV	Subsystem Functional Descriptions
V	Implementation Plan

VOLUME B - OSE FUNCTIONAL DESCRIPTION

I	OSE Objectives and Design Criteria
II	Design Characteristics and Restraints
III	STC System Level Functional Description
IV	STC and Subsystem OSE Equipment Groups Functional Descriptions
V	LCE Design Characteristics and Restraints
VI	LCE System Level Functional Description
VII	LCE Hardware Functional Description
VIII	AHSE Design Characteristics and Restraints
IX	AHSE System Level Functional Description
X	AHSE Hardware Functional Description
XI	MDE Requirements and Functional Description

VOLUME C - PROGRAM SYSTEM ANALYSIS

I	Introduction
II	Selection of Preferred System
III	Configurations
IV	Propulsion
V	Guidance and Control

1.0 INTRODUCTION. This volume contains the results of a systems study relating to the applicability of several types of propulsions systems for the VOYAGER Missions, completed by the General Electric Company in compliance with the JPL Statement of Work. The emphasis of the study was on overall Spacecraft System considerations affecting preferred propulsion systems for the VOYAGER Planetary Vehicle. In particular, the propulsion, configuration, and guidance and control subsystems of the spacecraft were considered in overall tradeoffs. The propulsion systems studied include:

a. A solid propellant engine used for orbit insertion, combined with a separate variable impulse multistart system for midcourse and orbit correction maneuvers.

b. The Apollo Lunar Excursion Module Descent Engine (LEMDE) system for all Planetary Vehicle maneuvers.

c. The Titan III C Transtage, used for orbit insertion, trajectory correction, and orbit trim maneuvers.

d. Modifications to the Titan III C Transtage configuration to reduce the overall length of the Planetary Vehicle.

Early in the study, it became evident that substantial modifications to all existing propulsion systems would be required in order to satisfy VOYAGER requirements. Consequently, the study was expanded to include a modified Apollo Lunar Excursion Module Descent Engine (LEMDE) system in the evaluation. This system employed LEMDE components (thrust chamber and tanks) repackaged into a more satisfactory configuration for the VOYAGER Planetary Vehicle.

The propulsion system selection study began by assembling design data for all the candidate propulsion systems. Conceptual spacecraft configurations were developed for each candidate. After several candidate configurations for each propulsion system had been identified, they were evaluated against mission requirements and the number of configurations reduced to one for each of the five system candidates mentioned earlier. Each of these five candidates were examined in more detail and the results assembled for a final tradeoff decision. These five candidate systems were then evaluated against the JPL mission constraints and competing characteristics in order to arrive at a final selection. The discussion of this final evaluation is discussed in Section 2.0 of this Volume. More detailed discussion of the configuration, propulsion, and guidance and control considerations is covered in Sections 3.0,4.0. and 5.0

The final selection was a solid propellant engine for orbit insertion and four hydrazine mono-propellant engines for midcourse and orbit adjust maneuvers. The spacecraft configuration for the preferred design is shown in Figure 1.0-1. The solid propellant engine recommended is a modification of the Wing 6 Minuteman Stage 2 engine. This engine is able to satisfy VOYAGER requirements with only relatively minor modification, specifically:

a. A reduction of cylindrical length of the case to reduce total impulse to VOYAGER requirements.

b. A change in the propellant burning rate to reduce thrust loading on the Planetary Vehicle.

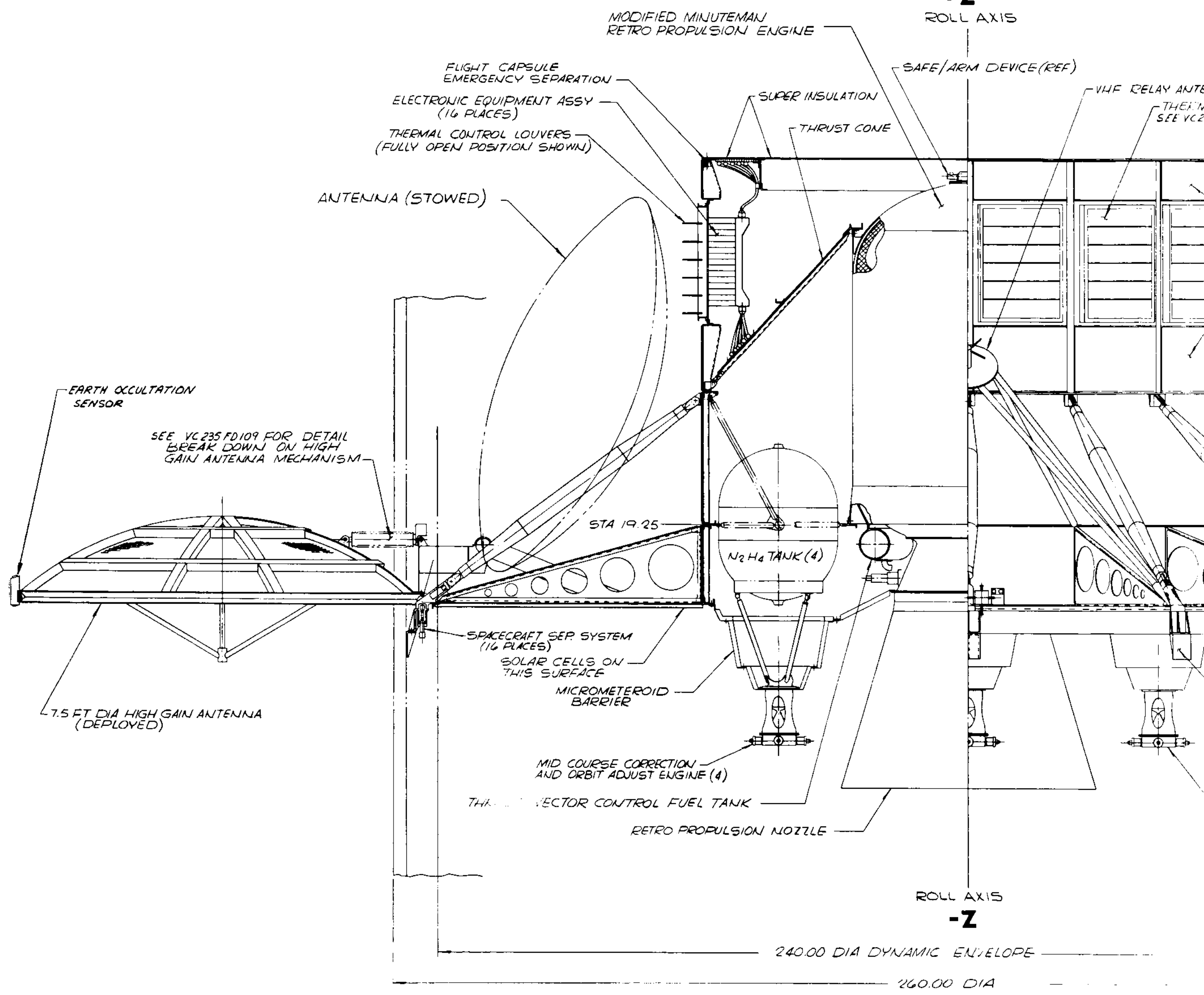
c. A reduction in nozzle throat diameter to accommodate the changed ballistic requirements.

In addition to these modifications to the main engine, the Minuteman Thrust Vector Control (TVC) system is modified by the substitution of a cold nitrogen pressurization system in lieu of a hot gas generator to pressurize the Freon tanks for increased long-life reliability.

The secondary propulsion system selected uses four hydrazine monopropellant engines. Shell catalyst is used to decompose hydrazine for the thrust chambers. Propellant acquisition in the four hydrazine tanks is accomplished with butyl rubber bladders. Thrust vector control of these four engines is accomplished with jet vanes in the thrust chamber exhaust. For maximum reliability, this system incorporates automatic shutdown of the opposite engine of a failed thrust chamber. This provides full mission capability with one engine pair not operating.

The predominant factor in the selection of this preferred propulsion system was overall spacecraft reliability. Two aspects of this were especially significant in the final decision. First, the attitude control of a spacecraft containing thousands of pounds of liquid propellant in partially full tanks has not been demonstrated using a low thrust level control system. Theoretical analysis of this problem is extremely difficult, but indicates that this may be a major problem in achieving mission success. A second major factor leading to this decision was that the modifications to the Minuteman Second Stage engine required to accomplish VOYAGER requirements were relatively simple and of the sort that would not be expected to degrade the reliability of a system specifically designed for reliable operation after long-term storage.

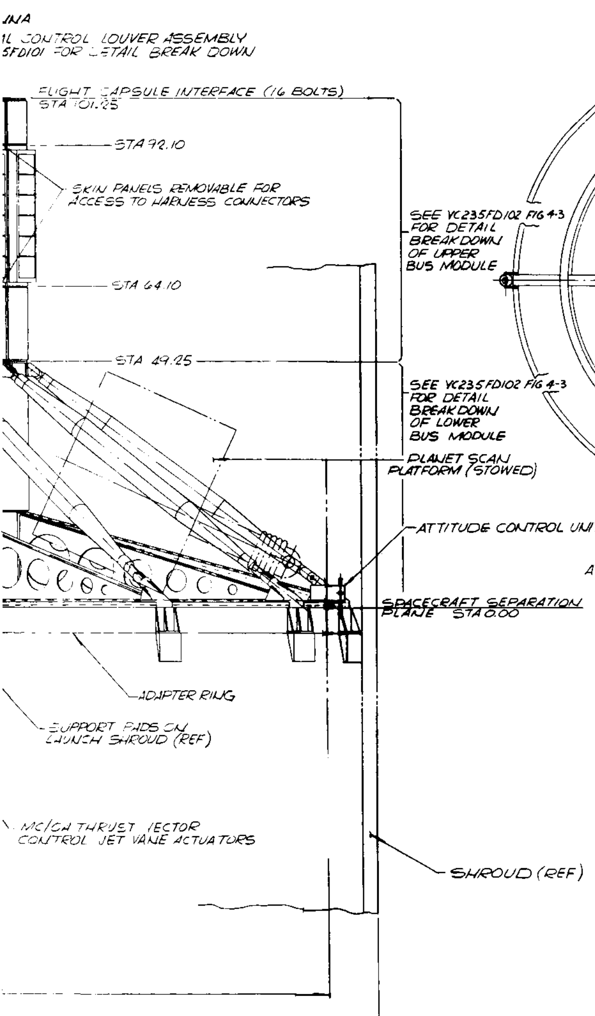
BAY NO.	
1	PC
2	PC
3	SC
4	SA
5	SC
6	SC
7	DA
8	DA
9	TE
10	CO

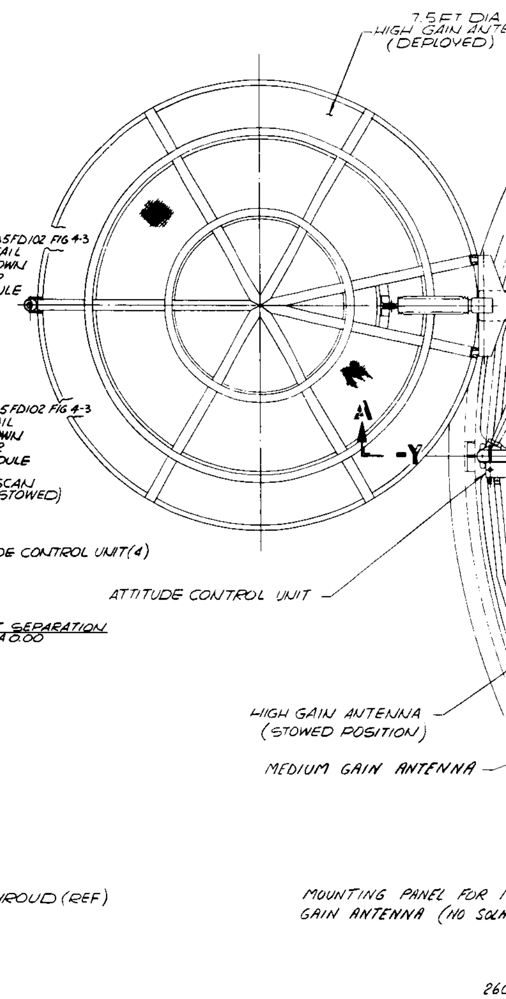


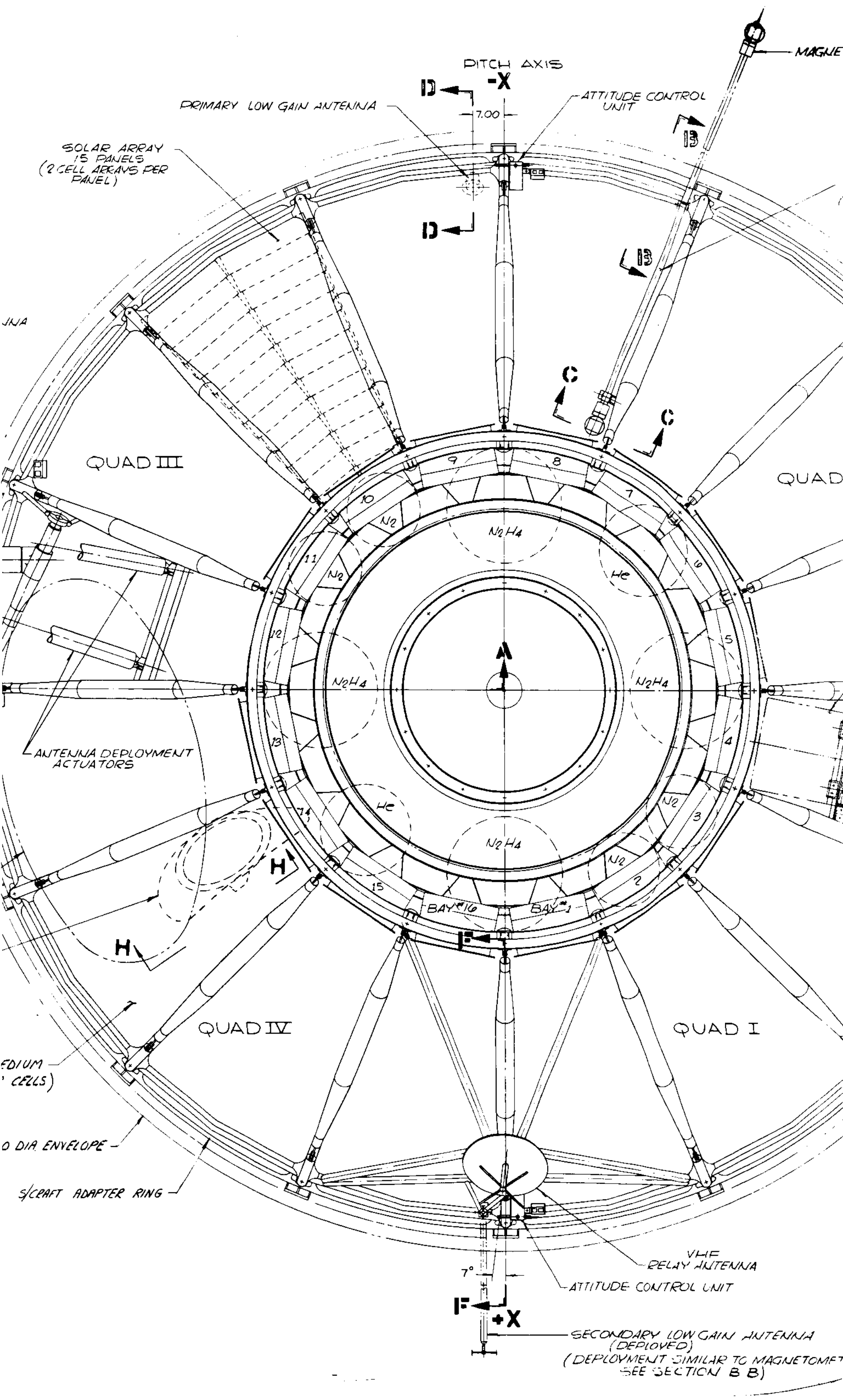
SECTION A-A

IBSYSTEM	BAY NO.	SUBSYSTEM
NER S/S	11	RADIO S/S
VER S/S	12	RADIO S/S
ENCE ELECTRONICS	13	RELAY RADIO
IRE	14	GUIDANCE & CONTROL S/S
NCE ELECTRONICS	15	GUIDANCE & CONTROL
NCE DAE	16	POWER S/S
A STORAGE		
A STORAGE		
EMETRY S/S		
MAND S/S		

TABLE SHOWING BAY ALLOCATIONS OF ELECTRONIC EQUIPMENT

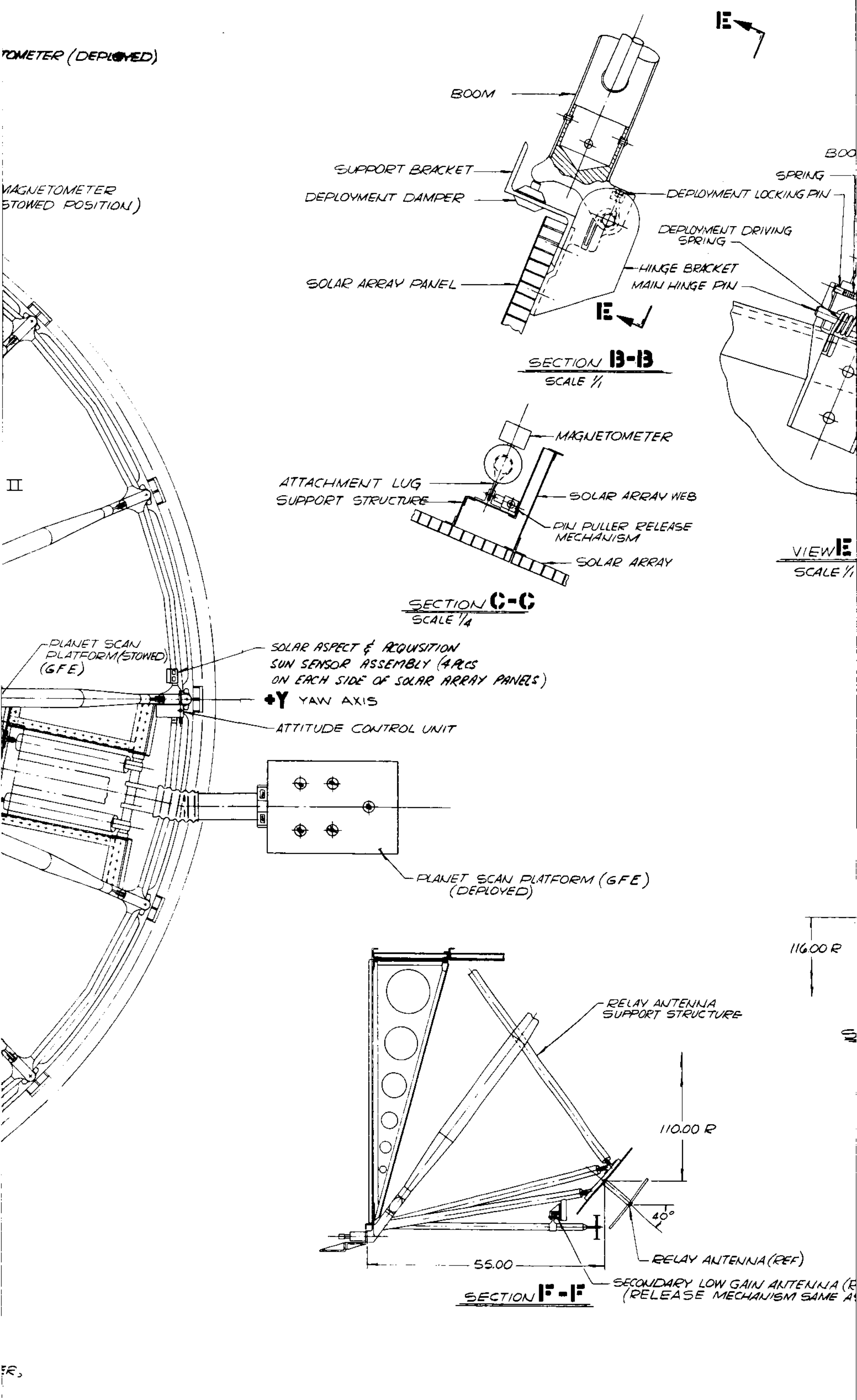






PITCH AXIS
-X
D
7.00
ATTITUDE CONTROL UNIT
PRIMARY LOW GAIN ANTENNA
SOLAR ARRAY 15 PANELS (2 CELL ARRAYS PER PANEL)
B
MAGNE
D
C
C
QUAD III
QUAD
N2H4
N2
He
A
N2H4
N2H4
He
N2H4
ANTENNA DEPLOYMENT ACTUATORS
H
H
BAY 16
BAY 1
F
QUAD IV
QUAD I
EDIUM
CELLS)
0 DIA ENVELOPE
S/CRAFT ADAPTER RING
VHF RELAY ANTENNA
ATTITUDE CONTROL UNIT
7°
F
+X
SECONDARY LOW GAIN ANTENNA (DEPLOYED)
(DEPLOYMENT SIMILAR TO MAGNETOME
SEE SECTION B B)

3



TOMETER (DEPLOYED)
MAGNETOMETER
STOWED POSITION)
II
BOOM
SUPPORT BRACKET
DEPLOYMENT DAMPER
SOLAR ARRAY PANEL
DEPLOYMENT LOCKING PIN
DEPLOYMENT DRIVING
SPRING
HINGE BRACKET
MAIN HINGE PIN
E
SECTION B-B
SCALE 1/1
BOO
SPRING
MAGNETOMETER
ATTACHMENT LUG
SUPPORT STRUCTURE
SOLAR ARRAY WEB
PIN PULLER RELEASE
MECHANISM
SOLAR ARRAY
SECTION C-C
SCALE 1/4
VIEW E
SCALE 1/1
PLANET SCAN
PLATFORM (STOWED)
(GFE)
SOLAR ASPECT & ACQUISITION
SUN SENSOR ASSEMBLY (4 PLCS
ON EACH SIDE OF SOLAR ARRAY PANELS)
+Y YAW AXIS
ATTITUDE CONTROL UNIT
PLANET SCAN PLATFORM (GFE)
(DEPLOYED)
116.00 R
RELAY ANTENNA
SUPPORT STRUCTURE
110.00 R
40°
RELAY ANTENNA (REF)
55.00
SECONDARY LOW GAIN ANTENNA (R
(RELEASE MECHANISM SAME A
SECTION F-F

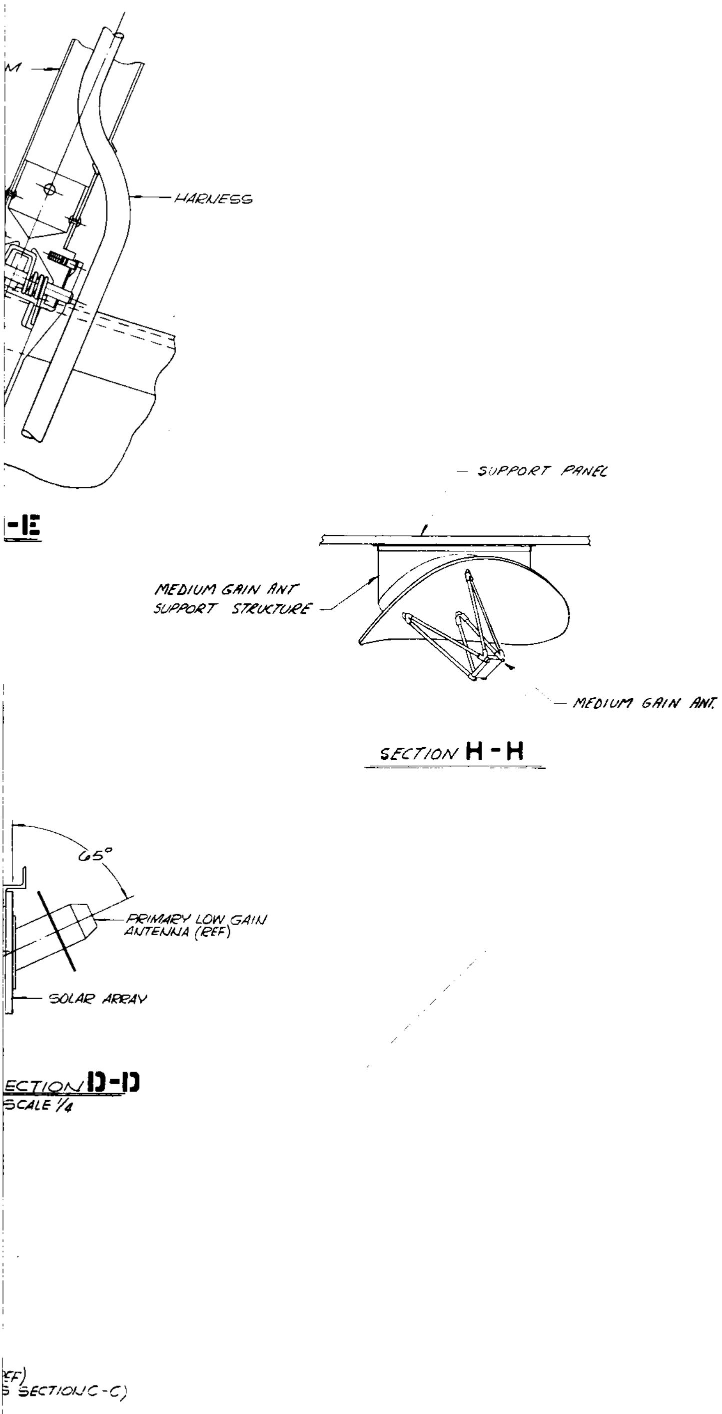


Figure 1.0-1. Spacecraft Configuration and General Arrangement

2.0 SELECTION OF PREFERRED SYSTEM. This section will discuss the factors considered in selecting a solid propellant orbit injection with a monopropellant midcourse and orbit adjust engines as the preferred design, in comparison with the other candidates listed in Section 1.0.

2.1 Description of Candidate Systems. The overall VOYAGER Mission requirements dictate a number of system level requirements and guide lines which are imposed upon the spacecraft configuration. These are summarized in Table 2.1-1. The constraints shown in this table are derived from JPL requirements, such as envelope restrictions. The guidelines imposed on configuration development are discussed in Section 2.0 of the Design Characteristics section of Volume A (VC220SR101).

All of the configurations described in this volume satisfy the configuration constraints, but some violate one or more of the guide lines of Table 2.1-1. There are two reasons for this. First, it is impractical to meet some guidelines with some propulsion system candidates. For example, a fixed solar array used with the LEMDE propulsion system unmodified is unattractive from a thermal balance standpoint. The second group of guideline rules not met are those in which configuration development was proceeding in parallel with the establishment of configuration guidelines. For example, a number of the configurations shown in Section 3.0 of this volume show a nondeployed high-gain antenna. These configurations were drawn prior to the system decision to select a deployed high-gain antenna and were prepared as an input to the study resulting in this decision. Each of the candidate configurations will be discussed in turn.

TABLE 2.1-1. VOYAGER CONFIGURATION REQUIREMENTS

Constraints

- Fit within dynamic envelope diameters and 208-inch length
- Support Flight Capsule at 120-inch diameter
- Withstand launch environment

Guidelines

- Capsule placed on shaded side of Planetary Vehicle
- One Planetary Scan Platform mounted at edge of solar array
- Minimize overall spacecraft length
- Approach uniform loads at capsule and shroud interfaces
- Electronic equipment mounted in a 16-side torus
- Fixed solar array mounted on 15 of 16 structural panels
- Mariner C antenna fixed in encounter position over a solar array structural panel
- Minimum spacecraft weight and inertias
- Deployed 7-1/2-foot high-gain antenna
- Propulsion system on longitudinal axis, nozzle toward sun in cruise attitude
- CG on longitudinal axis, far enough from propulsion system gimbal points to assure autopilot control
- Modular construction

2.1.1 Modified Minuteman Configuration. A top isometric view of this configuration is shown in Figure 2.1-1, illustrating the placement of major spacecraft elements. An exploded view of the same configuration is shown in Figure 2.1-2. This view illustrates the structural configuration and the internal placement of propulsion and attitude control elements. The basic structure is a 120-inch diameter cylindrical shell with 16 longerons. Rings are provided for two manufacturing joints which divide the structure into three basic modules; the upper module contains electronic equipment, the middle module contains the midcourse and orbit adjust propulsion system, and the lower module supports the solar array and the attitude control cold gas jet subsystem. The retropropulsion engine is supported at its forward end by a monocoque structure which is part of the electronic module and serves both as a structural support for the

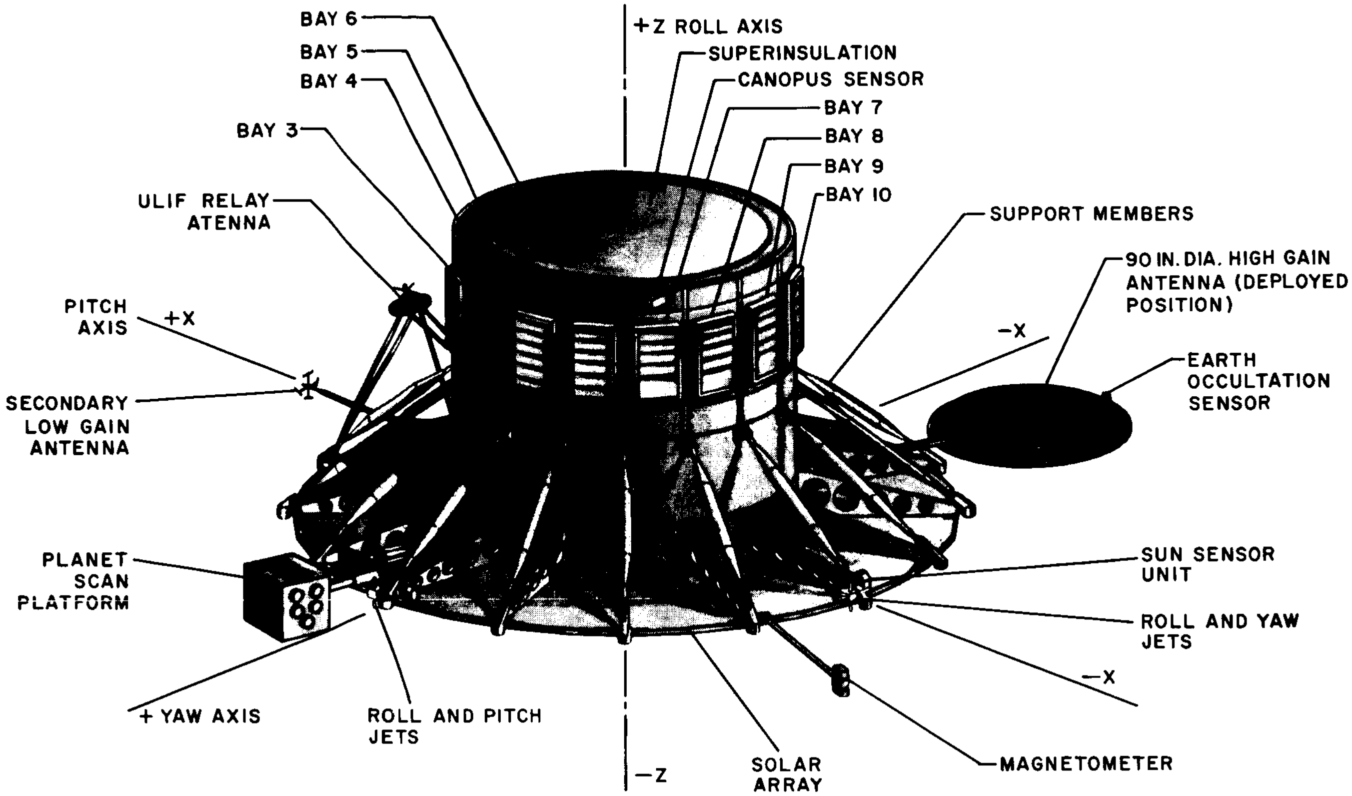
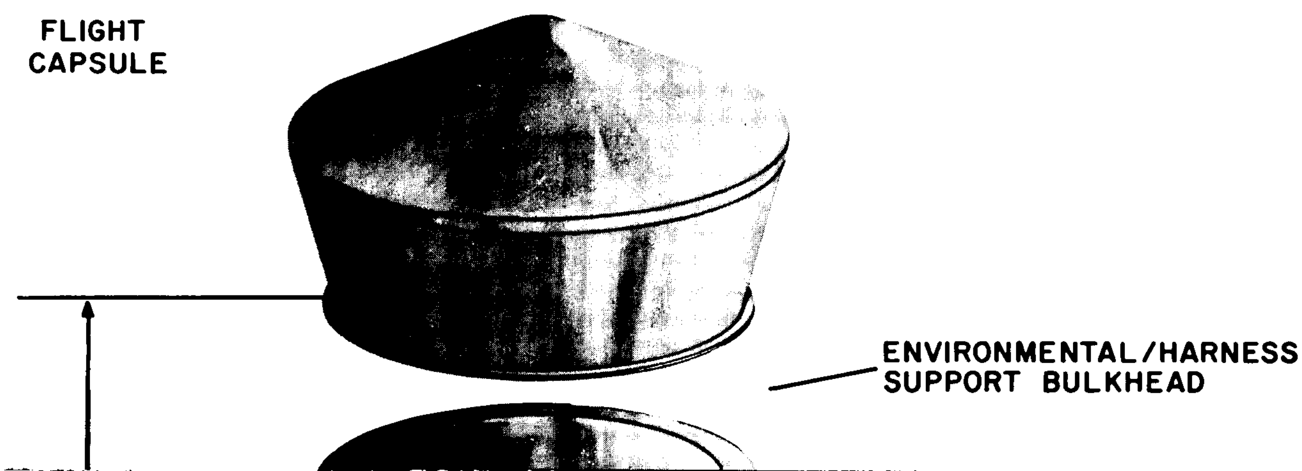
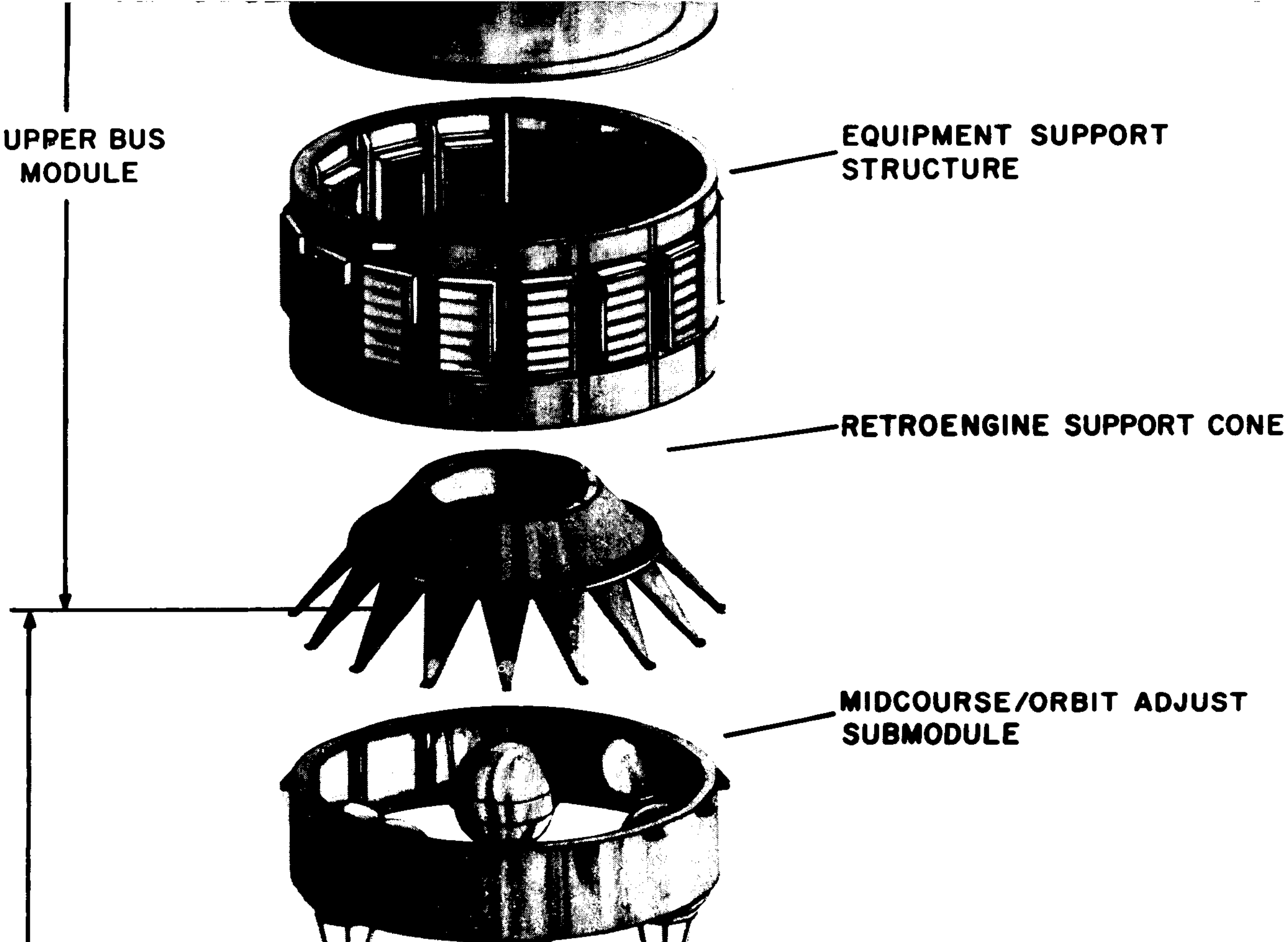


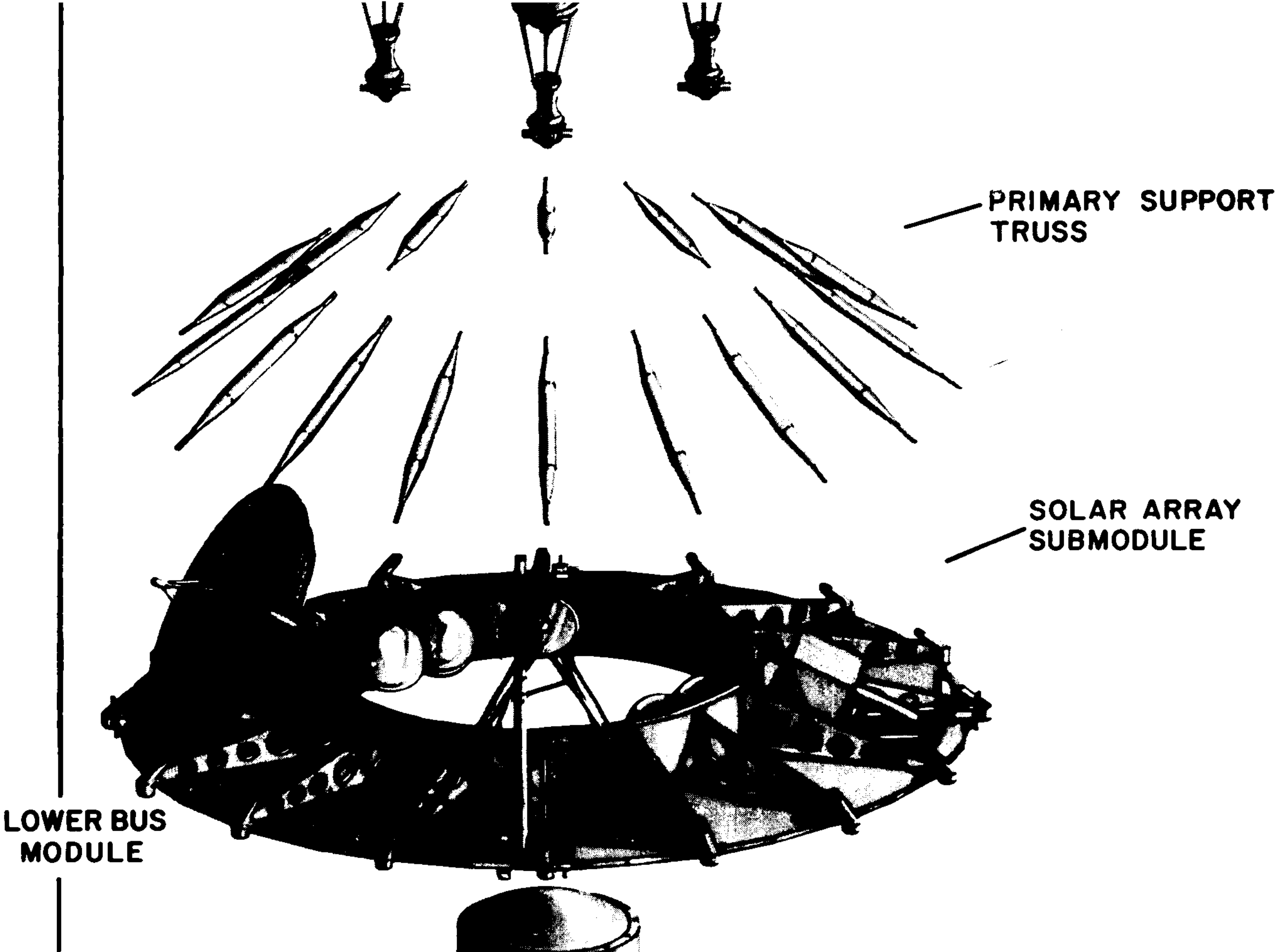
Figure 2.1-1. Spacecraft Equipment Arrangement (Top View)

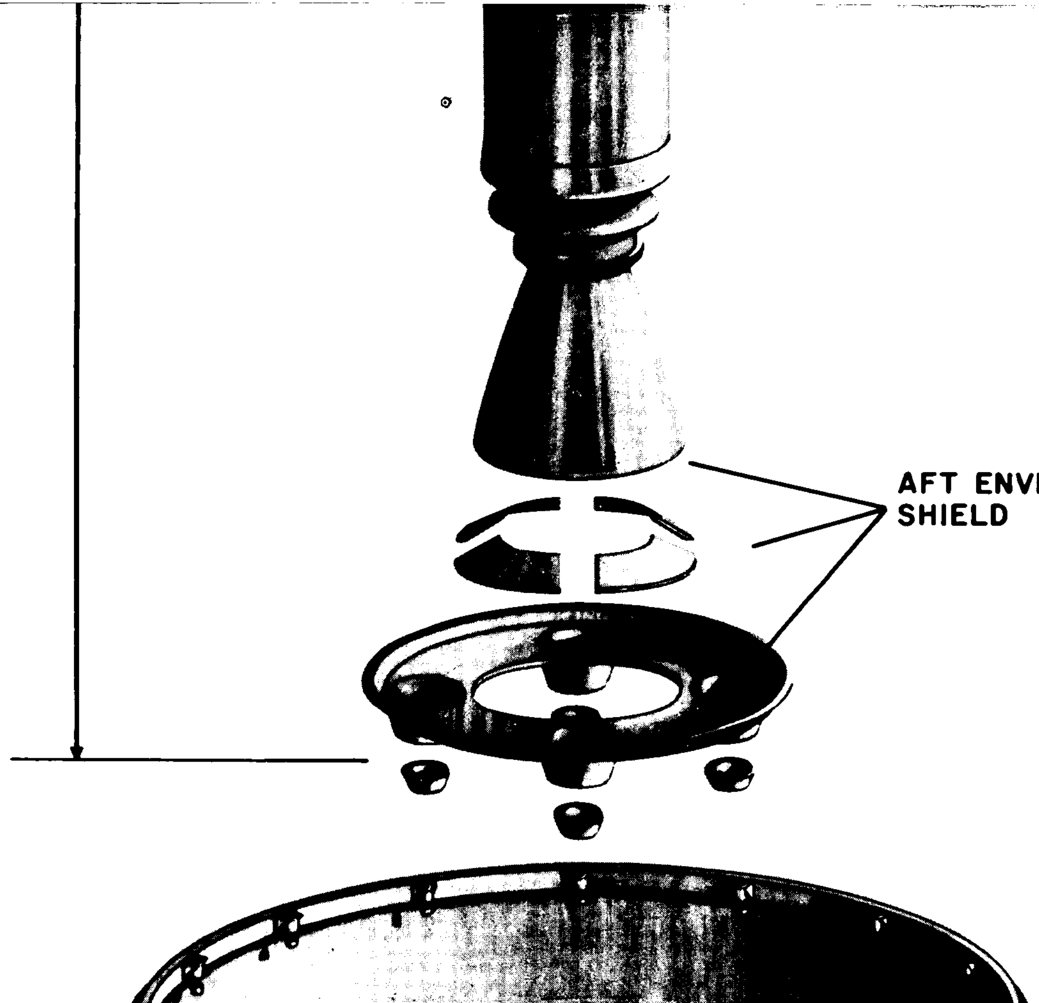
retropropulsion system and the harness tray for the electronic equipment module. The spacecraft is supported at the launch vehicle interface by fittings at the end of the solar array. Launch loads are transmitted to the spacecraft through 16 support tubes which connect the outer end of the solar array to the bottom of the electronic module and the retropropulsion engine support cone. The propulsion system description has been summarized in Section 1.0 and will be discussed in more detail in Section 4.0. Environmental shields are provided at both ends of the spacecraft to provide meteoriod protection and thermal insulation.

2.1.2 Unmodified LEMDE. An isometric view of the unmodified LEMDE configuration is shown in Figure 2.1-3. Because of the packaging arrangement of the unmodified LEMDE propulsion system and the corresponding structural arrangement, this configuration is markedly different from all other candidates. The propulsion system tankage occupies a large fraction of the available area in the shroud envelope, so the unmodified LEMDE configuration was forced to resort to deployed solar panels for the power subsystem. There was insufficient area between the shroud and the LEM structure to provide the required area, and it was not feasible to place solar cells on the LEM structure, since they would become overheated during the early part of the mission. Figure 2.1-4 is an exploded view of the unmodified LEMDE configuration. The most convenient way of structurally supporting the LEMDE propulsion system proved to be a conical shell attached at the larger diameter to the Saturn 5 shroud and supporting the flight capsule at the upper end. The eight upper landing gear fittings of the LEMDE propulsion system are fastened to eight longerons stiffening this honeycomb conical shell. The existing LEMDE structure is unmodified except for deletion of fittings and backup structure for unused items, such as water tanks. It is shrouded with a thermal blanket and meteoroid shield in a manner similar to the existing LEMDE vehicle.


FLIGHT
CAPSULE
ENVIRONMENTAL/HARNESS
SUPPORT BULKHEAD


UPPER BUS
MODULE
EQUIPMENT SUPPORT
STRUCTURE
RETROENGINE SUPPORT CONE
MIDCOURSE/ORBIT ADJUST
SUBMODULE


PRIMARY SUPPORT TRUSS
SOLAR ARRAY SUBMODULE
LOWER BUS MODULE



AFT ENVIRONMENTAL
SHIELD

4

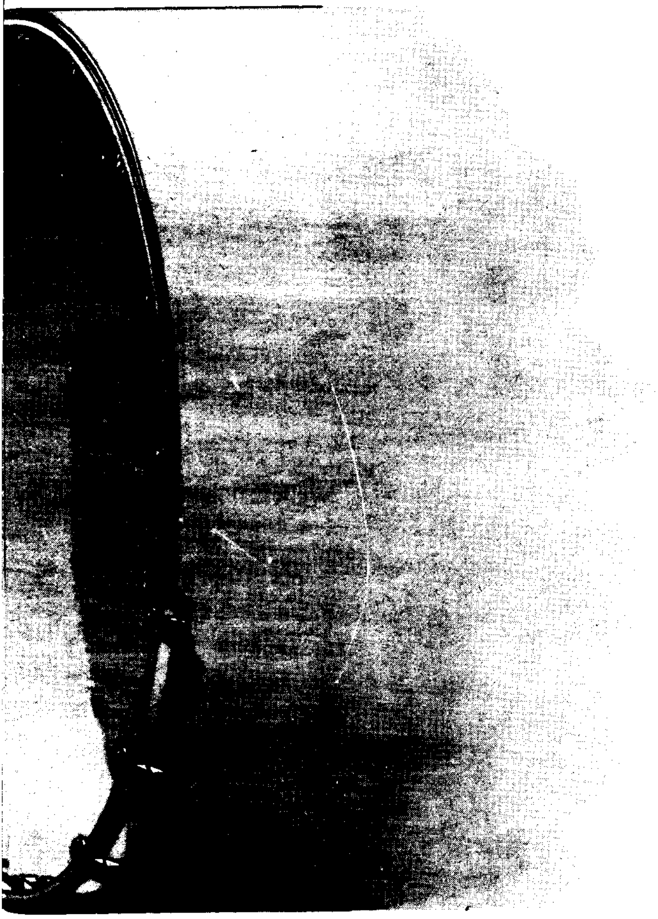


Figure 2.1-2. Spacecraft Structural
Breakdown - Preffered Design

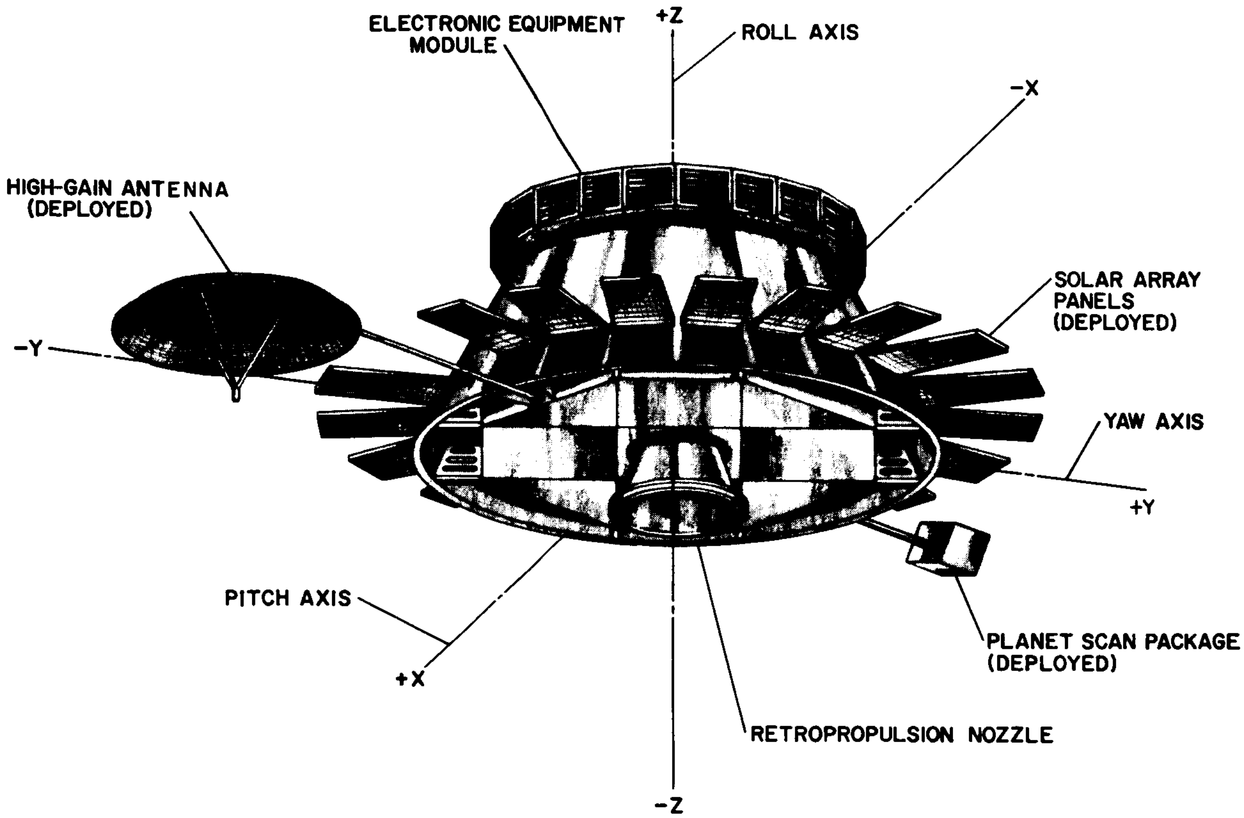


Figure 2.1-3. Unmodified LEMDE Configuration

The electronic equipment of the spacecraft is contained in a 20-side torodial ring mounted near the top of the honeycomb support cone. Deployable solar array panels are attached near the bottom of the cone with gaps in the array to permit deployment of the high-gain antenna and planetary scan platform.

The LEM descent propulsion system considered in the final tradeoff utilized the main thrust chamber for all propulsive maneuvers. Orbit insertion was accomplished by using the full 10,500-pound thrust capability of the propulsion system. This yields a maximum accleration during orbit insertion only slightly in excess of 1g, which is desirable from the spacecraft structural standpoint. Midcourse and orbit adjustment maneuvers are accomplished at the 1000-pound thrust level, the minimum capability of this system. This results in a minimum delta V for midcourse corrections of about a quarter of a meter per second, which is slightly higher than desired. However, the mission can be accomplished readily with this minimum impulse capability. Although the single chamber propulsion system provides no capability for roll control, it is expected that roll control during engine firing could be provided by the spacecraft attitude control system, and this is the system which was evaluated during the final tradeoffs.

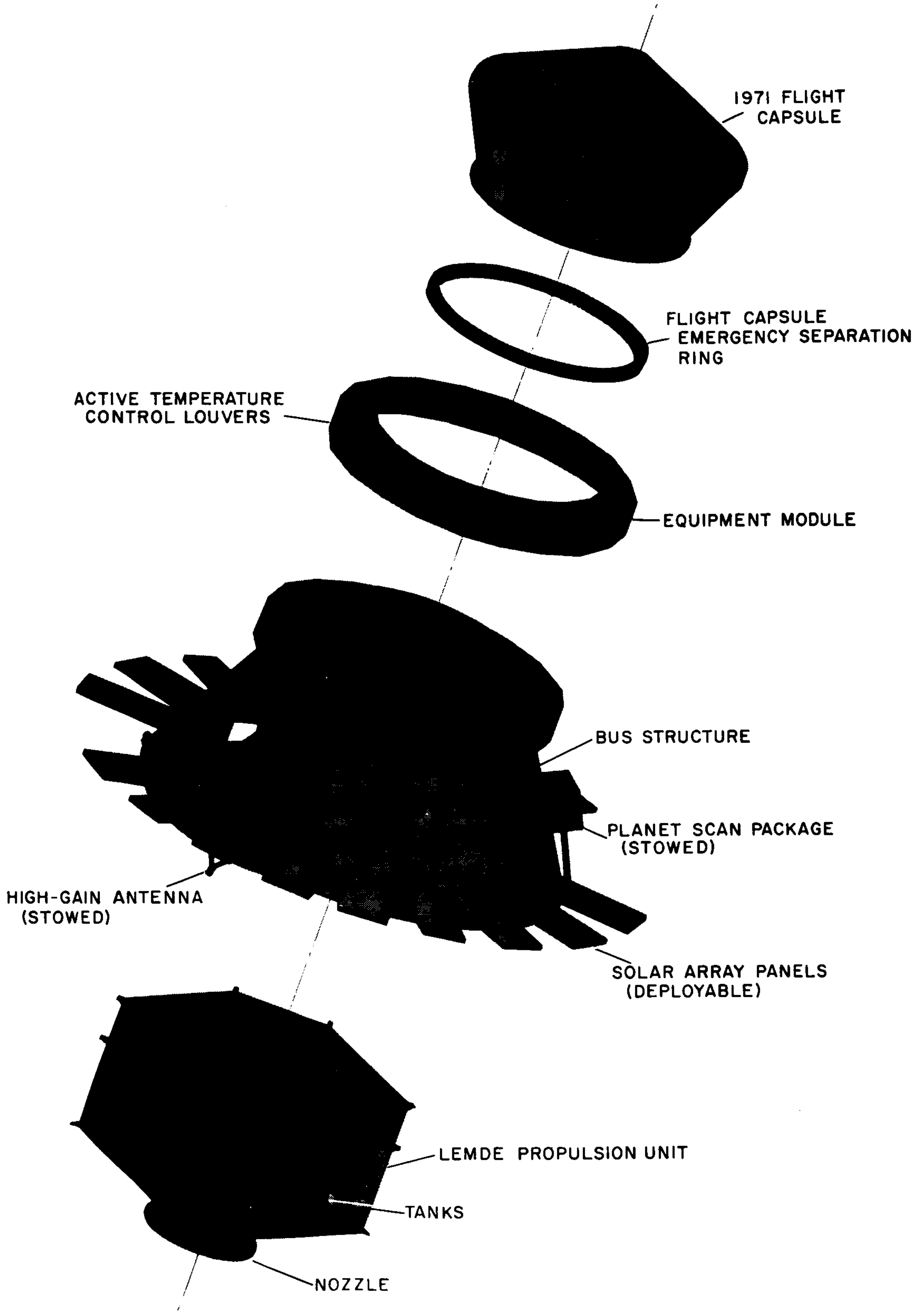


Figure 2.1-4. Unmodified LEMDE Configuration of Exploded View

Several modifications to the existing LEM descent propulsion system would be required in order to satisfy VOYAGER Mission requirements. In the LEM application of this system, propellant acquisition is provided by operation of the LEM ascent attitude control engines to provide propellant settling. Since this technique is not applicable to VOYAGER, some modification would have to be made to the propulsion system to assure propellant acquisition under zero g.

The current design of the LEM descent propulsion stage uses cryogenic storage for the pressurizing helium gas. Such a system would not be suitable for long-trip time required for the VOYAGER Mission. Consequently, a modification of the standard LEM to incorporate an alternate pressurization system, such as high pressure helium gas, would be required. The present actuators of the LEM descent stage have inadequate control response for the VOYAGER Spacecraft. They would have to be replaced with faster-acting mechanisms. Finally, all configurations available within the 208-inch shroud length restriction that utilize the standard LEM descent propulsion system result in center-of-gravity to engine-gimbal points which are too close together to permit effective control by the spacecraft auto-pilot. To overcome this problem, it would be necessary to relocate the engine further aft for all unmodified LEM descent configurations.

2.1.3 Modified LEMDE. Because of the basic capability of the LEM descent propulsion system to perform all propulsive maneuvers for the Planetary Vehicle with a single thrust chamber, this is an attractive system candidate. However, the propulsion system arrangement applicable to the LEM vehicle is not well adapted to the VOYAGER requirement. Furthermore, because of the requirement that the LEM descent propulsion structure withstand extreme landing loads when LEM touches down on the moon, the structural weight is very large compared to VOYAGER requirements. In order to get a more thorough evaluation of the LEM descent propulsion system as applied to VOYAGER, it was decided to study a reconfigured version of this propulsion system, since it was already clear that the basic LEM system could not be used without substantial modification. This modified configuration is illustrated in Figure 2.1-5. The details of this arrangement can be seen much better in Figure 2.1-6, an exploded view of the modified descent LEM propulsion system as applied to a VOYAGER Spacecraft.

In this design the components of the LEM propulsion system have been repackaged in a more compact arrangement. The propellant tanks have been modified by removing the cylindrical portion to create four spherical tanks. These are mounted to the honeycomb "eggcups" as in the unmodified LEMDE configuration. The basic propulsion structure is a 120-inch diameter circular shell with one main cross beam and two auxiliary beams perpendicular to the main center-line beam. These beams support both the modified propellant tanks and the unmodified LEMDE engine. Two high-pressure spherical helium tanks complete the propulsion assembly. Also included in this central propulsion module are the four 16-inch diameter spherical nitrogen tanks for the attitude control subsystem. The electronic equipment of the spacecraft is packaged as in the preferred design described in Volume A, in 16 bays attached to a semi-monocoque structure with 16 stringers to carry the mounting loads of the electronic assemblies and support the Flight capsule interface ring.

This configuration uses a 16-panel solar array like the preferred configuration selected. Thermal insulation and meteoroid protection for the spacecraft is provided by a segmented

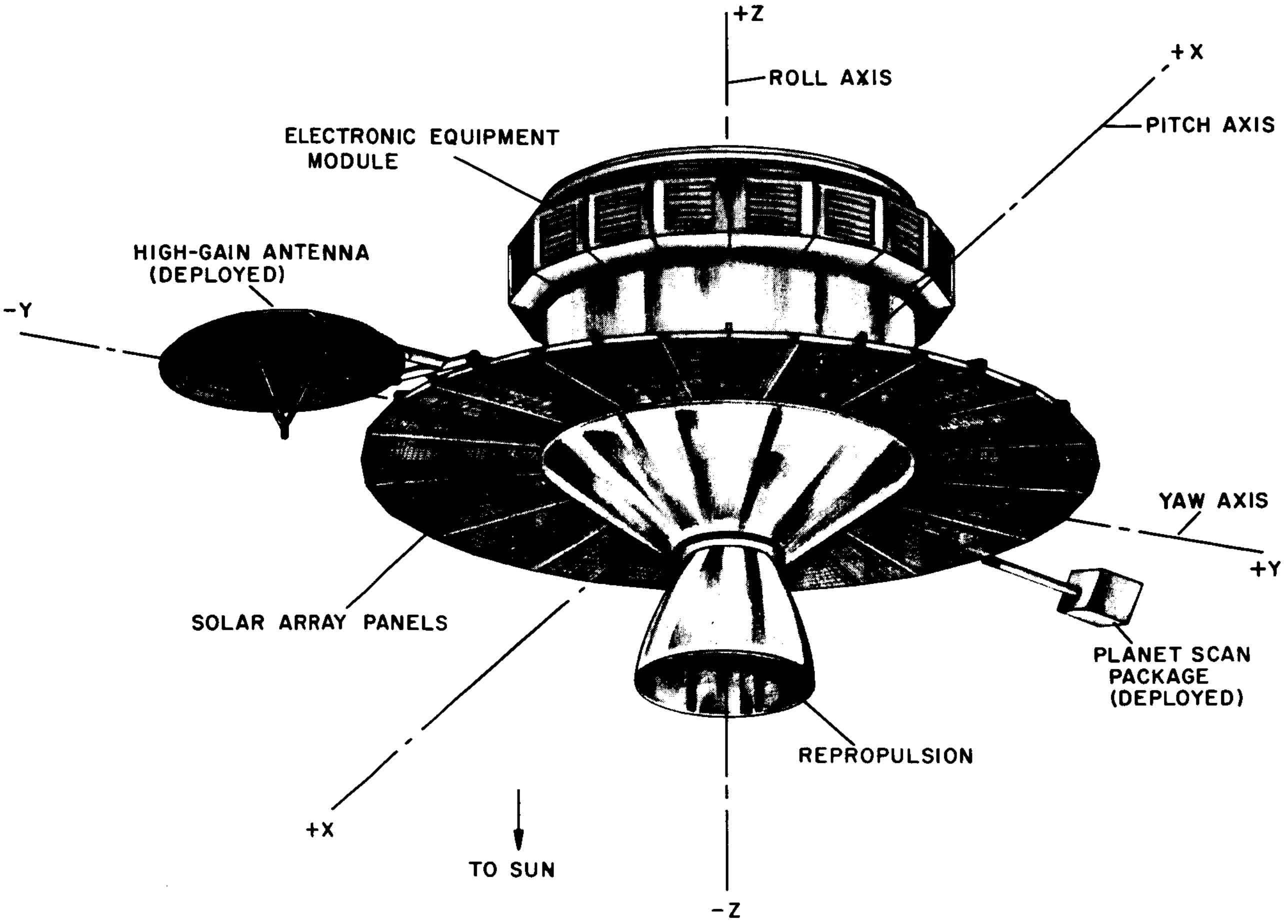


Figure 2.1-5. Modified LEMDE Configuration

cone enclosing the bottom of the propulsion system, an insulating disc covering the Flight Spacecraft adapter ring, and thermal insulation on the internal surfaces of the cylindrical body of the spacecraft. This spacecraft is supported from the Saturn shroud by a uniformly loaded honeycomb core. Separation is accomplished by severing the upper end of this cone with encapsulated MDF.

2.1.4 Unmodified Transtage Configuration. An isometric view of this configuration is shown in Figure 2.1-7. Again, the exploded view, Figure 2.1-8, reveals more of the Planetary Vehicle detail. In this configuration, the propulsion system module configuration and structure of the Titan III C transtage is retained. The control module of the Titan III C transtage has been removed, since it is not applicable to the VOYAGER requirements. In order to satisfy the VOYAGER long-life requirement, several modifications to the standard Transtage components are required, such as pressurizing the engine gimbaling actuators to prevent loss of hydraulic fluid during a long space storage. Because of the large thrust available from the two 8,000-pound thrust chambers, the minimum delta V available from the unmodified Transtage is too large to satisfy the midcourse correction requirements of the VOYAGER Mission. To overcome this limitation, four-gimbaled bipropellant thrust chambers

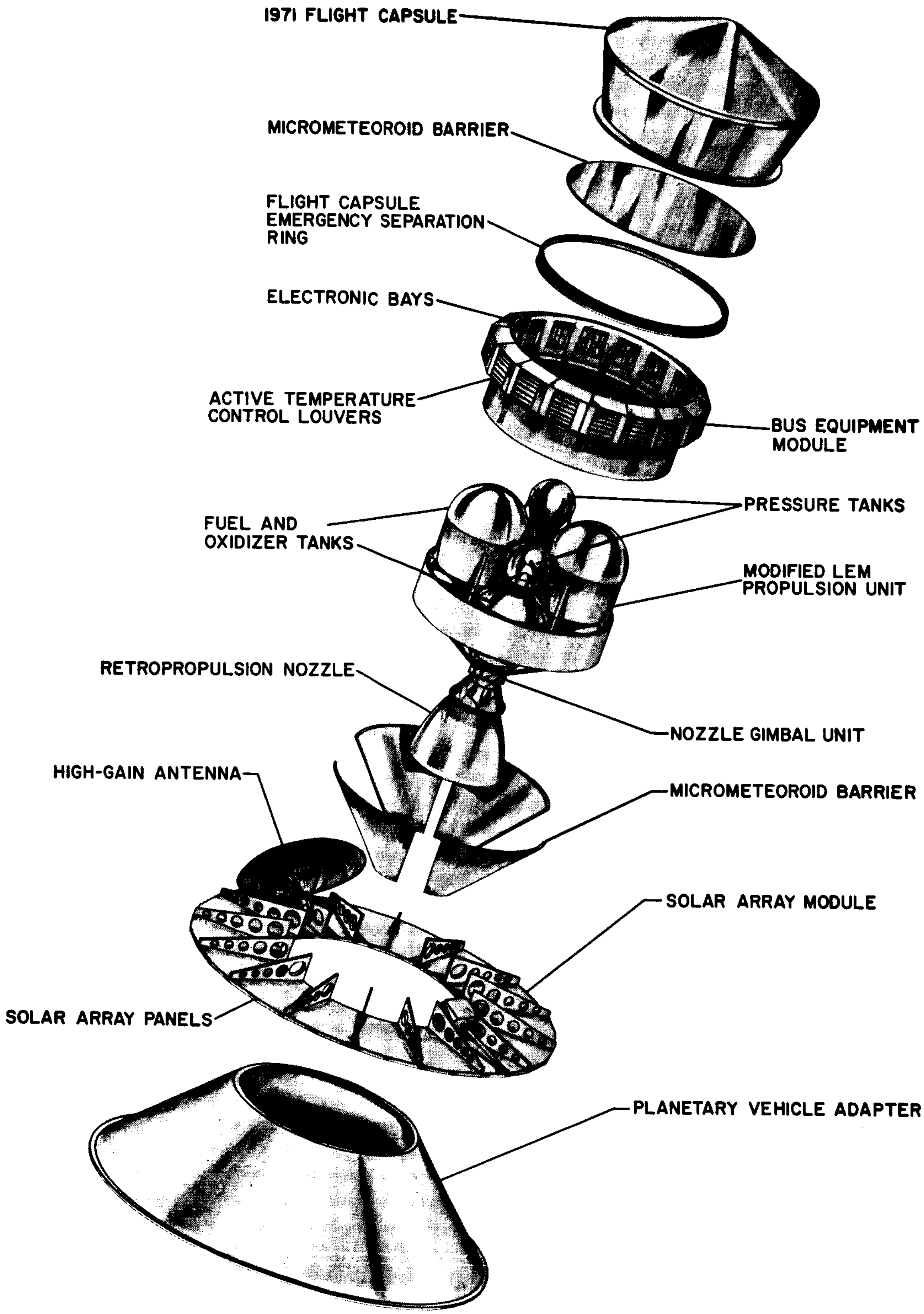


Figure 2.1-6. Modified LEMDE Configuration, Exploded View

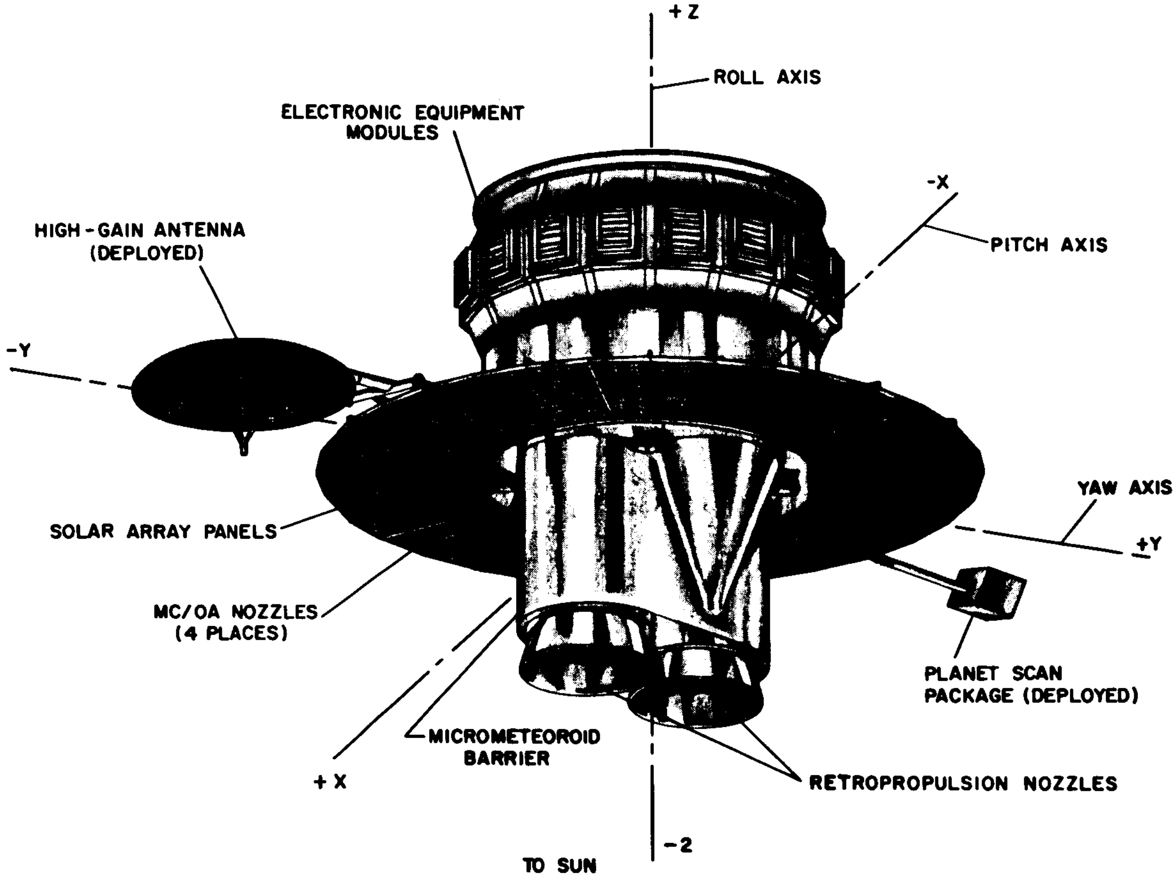


Figure 2.1-7. Titan Transtage Configuration

have been added to the basic Transtage propulsion module. These engines are the pitch and yaw engines of the basic Transtage control module. In the VOYAGER application they withdraw fuel from the main propellant tanks instead of from separate tanks, as in the standard Transtage configuration. It is planned that large-velocity corrections would be made with the main Transtage thrust chambers, and only small corrections would utilize the four vernier chambers, since they are limited in burn-time duration. It is not planned to operate both main and vernier engines at the same time in order to simplify autopilot requirements.

The tanks of the standard Transtage configuration are able to contain approximately twice as much propellant as is required to accomplish the VOYAGER Mission. This permits a significant simplification of the Transtage pressurization system with resulting enhanced reliability. A major fraction of the required pressurization gas is contained in the ullage volume of the half-empty propellant tanks. There is almost enough gas contained in this volume to permit simple blowdown operation of the Transtage. However, the amount of gas available is so marginal for this purpose that substantial testing would be required to verify simple blowdown operation of this propulsion system. Hence, two additional pressurization

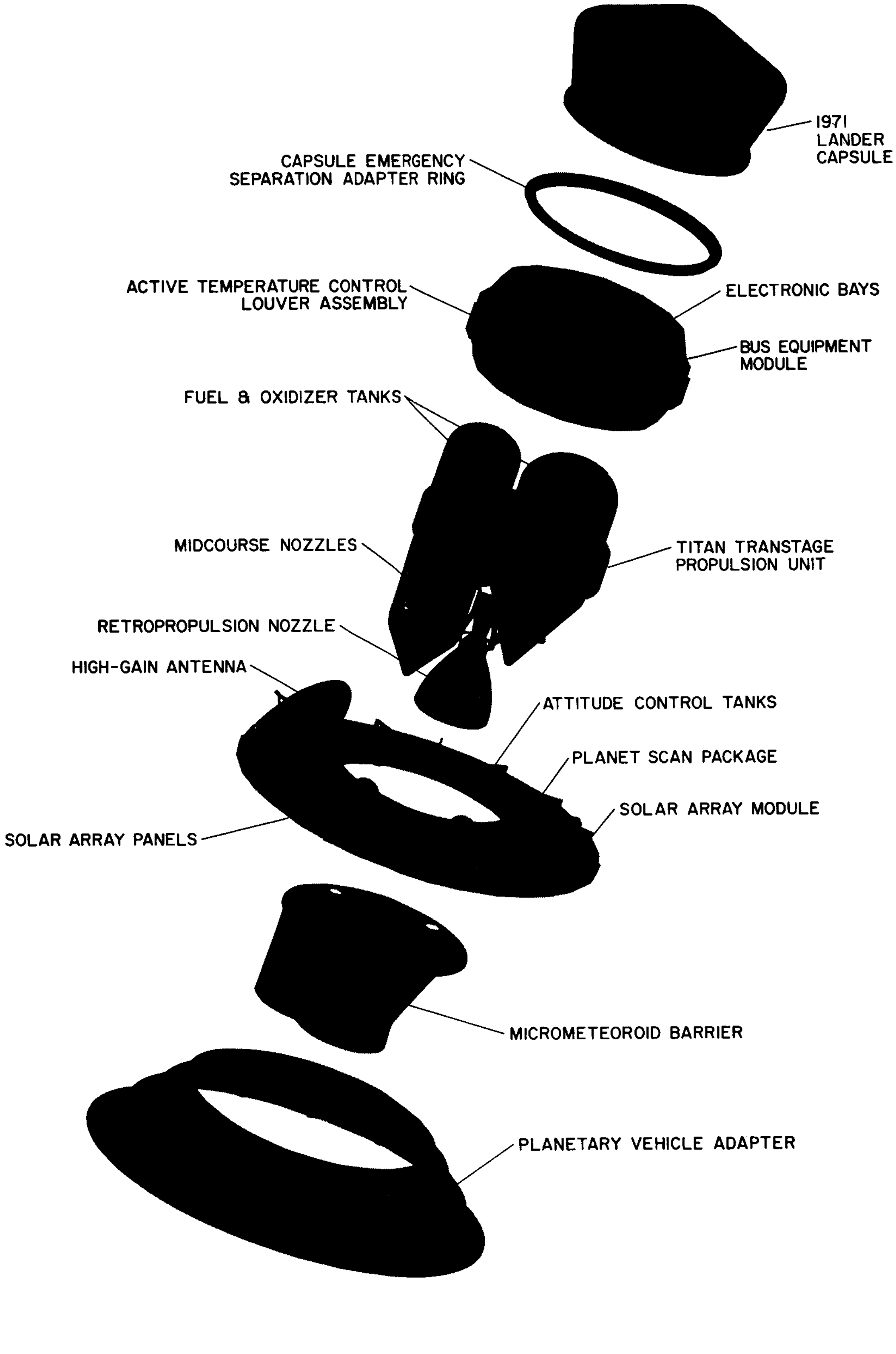


Figure 2.1-8. Titan Transtage Configuration, Exploded View

tanks are included in the propulsion module. The midcourse correction maneuvers for the VOYAGER Mission will be accomplished by using the blowdown capability of the gas stored in the main tanks. Then, during main retropropulsion firing, as the tank pressure falls below a predetermined minimum, a squib valve will be opened, admitting pressurization gas from the auxiliary tanks into the main tank. The flow of this gas can be regulated with orifices, thereby eliminating regulators from the pressurization system and providing a significant improvement in propulsion system reliability.

The spacecraft configuration, other than the propulsion system, is very similar to the modified LEM descent configuration just described. Electronic equipment is mounted on a 16-bay torus above the mounting ring of the propulsion module and enclosing the upper part of the propellant tanks. A fixed solar array is used. This array supports the high-gain antenna and planetary scan platform, as in the preferred configuration. Thermal insulation and micrometeorite protection for the aft portion of the Transtage tanks are provided by a contoured super-insulation shield. The spacecraft is supported on the Saturn 5 Launch Vehicle by a conical honeycomb adapter similar to the one used with the modified LEM descent configuration. To minimize the difficulty involved in separating the spacecraft caused by the close clearance between the thrust chamber nozzles and the 10-foot-diameter adapter used in the LEM descent system, the Transtage adapter has been enlarged so that it is attached part way out along the solar array. The loads introduced at this point are carried into the main cylindrical body of the spacecraft by short thrust tubes which connect the adapter interface diameter to the main body.

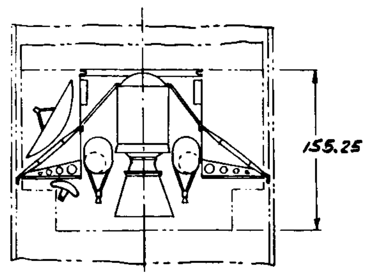
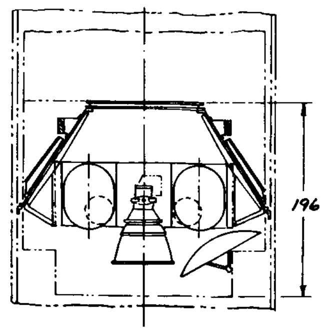
2.1.5 Modified Transtage Configuration. The only difference in the Modified Transtage Configuration considered in the final tradeoffs is that 15 inches of cylindrical tank has been removed. The principal effect of shortening propellant tanks is to reduce the amount of ullage available to containing pressurizing gas. This requires that the auxiliary pressurizing spheres contain more helium stored at higher pressure and released into the main tanks earlier in the retropropulsion system firing. The principal advantages of this configuration are a 15-inch length reduction of the overall Planetary Vehicle and some weight reduction in the spacecraft structure.

Control of propellant slosh and acquisition for the Transtage will require some development work. Acquisition of propellant in the present designs is accomplished by a combination of screens in the bottom of the propellant tanks, plus settling forces applied by the control module engines prior to main engine start. The configuration considered in the final tradeoff among candidate systems used a series of propellant control screens to assure positive propellant acquisition for engine starting and to minimize sloshing of the propellant in the half-empty Transtage tanks. Although this method of propellant control seems reasonable, it has not been demonstrated under months of zero-g storage. Consequently, there must remain some question about its feasibility.

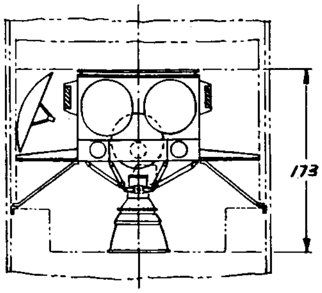
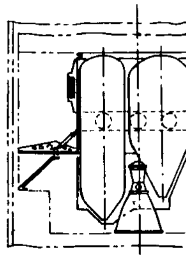
Some of the pertinent configuration information for the five propulsion candidates has been assembled in the matrix of Figure 2.1-9. From this figure, it can be seen that the adopted modified Minuteman configuration is appreciably lighter than any of the other candidates considered. It is also the shortest of the candidate systems. Moreover, the selected configuration is competitive with the other configurations in terms of desirable mass properties and dynamic response.

PROPULSION SELECTION MATRIX

CANDIDATE SYSTEMS INFORMATION

	MODIFIED MINUTEMAN CONFIGURATION "G"	L.E.M.D.E. C UNMODIFIED "H"
		
TOTAL EARTH TAKE-OFF WEIGHT *	17,289 #	20,205 #
ADAPTER- S/C TO BOOSTER WEIGHT	113 #	326 #
TOTAL CRUISE WEIGHT	17,176 #	19,879 #
BURN-OUT WEIGHT	4,016 #	5,578 #
PROPULSION SYS. TOTAL WEIGHT - EMPTY	1,617 #	3,019 #
PROPULSION SYS. TOTAL WEIGHT - LOADED	11,863 #	14,320 #
BUS WEIGHT	2,312 #	2,519 #
ENVIRONMENTAL SHIELD WEIGHT	70 #	40 #
SOLAR ARRAY AREA - FIXED	238 ft²	NONE
SOLAR ARRAY AREA - DEPLOYED	NONE	229 ft²
RECOMMENDED SEPARATION SYSTEM	GAS THRUSTER	MILD DETONATING FUSE
DIM.-NOZZLE EXIT TO SOLAR ARRAY	38"	MINIMAL
NATURAL FREQUENCY- FIRST MODE. LONGITUDINAL	22.5 C.P.S.	21.3 C.P.S.
NATURAL FREQUENCY- FIRST MODE. LATERAL	13.0 C.P.S.	12.5 C.P.S.
NATURAL FREQUENCY- FIRST MODE. TORSIONAL	15.2 C.P.S.	19.4 C.P.S.
SHROUD LENGTH L_1	562.50"	644.00"
SHROUD LENGTH L_2	323.25"	364.00"
I_X PREMATURE CAPSULE SEPARATION	3,773 SLUG-ft²	7,948 SLUG-ft²
I_X AFTER ORBIT INJECTION	14,499	17,451
I_X AFTER CAPSULE SEPARATION	3,634	3,906
I_Y PREMATURE CAPSULE SEPARATION	4,179	5,382
I_Y AFTER ORBIT INJECTION	14,093	16,324
I_Y AFTER CAPSULE SEPARATION	2,781	2,459
I_Z PREMATURE CAPSULE SEPARATION	3,925	9,966
I_Z AFTER ORBIT INJECTION	6,997	7,448
I_Z AFTER CAPSULE SEPARATION	3,970 SLUG-ft²	3,848 SLUG-ft²
TOTAL A/C IMPULSE ESTIMATE (LB.-SEC.)	1730	2330
TOTAL Δ C.G. DUE TO ENGINE FIRING — ORBIT INJECTION X	.25 Δ INCHES	.03 Δ INCHES
TOTAL Δ C.G. DUE TO ENGINE FIRING — ORBIT INJECTION Y	.06	.01
TOTAL Δ C.G. DUE TO ENGINE FIRING — ORBIT INJECTION Z	26.03	46.95
TOTAL Δ C.G. DUE TO ENGINE FIRING — OA X	.13	0
TOTAL Δ C.G. DUE TO ENGINE FIRING — OA Y	.18	05
TOTAL Δ C.G. DUE TO ENGINE FIRING — OA Z	1.05 Δ INCHES	1.48 Δ INCHES
MINIMUM DISTANCE - C.G. FROM GIMBAL POINT	36.40"	2.99"

* FOR ONE PLANETARY VEHICLE.

	ONFIGURATIONS	TRANS
	MODIFIED "L"	UNMODIFIE
		
	17,638 #	19,350 #
	307 #	357 #
	17,331 #	18,993 #
	4,400 #	4,880 #
	1,984 #	2,386 #
	11,915 #	13,605
	2,376 #	2,388 #
	40 #	106 #
	218 ft²	236 f
	NONE	NONE
	MILD DETONATING FUSE	GAS THRUS
	92"	80"
	12.4 C.P.S.	16.0 C.P.
	9.8 C.P.S.	13.0 C.P.
	18.0 C.P.S.	18.0 C.P.S
	598.00"	606.00"
	341.00"	345.00
	4,946 SLUG-ft²	10,951 S
	13,309	16,599
	3,495	6,962
	3,550	8,479
	12,383	15,775
	2,249	5,810
	5,567	5,163
	7,309	7,242
	3,709 SLUG-ft²	3,649 S
	1510	2550
	.04 Δ INCHES	.20 Δ
	.01	1.51
	26.88	58.46
	.01	.04
	.07	.21
	.12 Δ INCHES	3.39 Δ
	58.30"	52.70"

✓

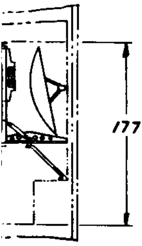
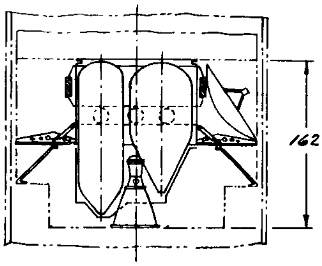
…STAGE CONFIGURATIONS	
…D "N"	MODIFIED "Q"
	
	18,233 #
	357 #
	17,876 #
	4,753 #
	2,366 #
	12,396 #
	2,387 #
	93 #
…2	236 ft2
	NONE
…TER	GAS THRUSTER
	80"
…S.	17.0 C.P.S.
….	15.0 C.P.S.
	19.0 C.P.S.
	576.00"
	330.00"
…LUG - ft²	4,806 SLUG - ft²
	14,255
	3,629
	3,127
	13,737
	2,717
	5,086
	7,667
…JG - ft²	4,138 SLUG - ft²
	1600
…NCHES	.57 Δ INCHES
	.03
	30.81
	.11
	.09
…ICHES	.29 Δ INCHES
	31.60"

Figure 2.1-9. Propulsion Selection Matrix

2.2 Evaluation Against Mission Constraints. After the five candidate systems had been selected and sufficient information was available, each of these candidates were evaluated against their ability to satisfy the mission constraints of planetary quarantine and minimum schedule risk.

2.2.1 Planetary Quarantine. It is a firm program requirement that contamination of Mars by viable terrestrial organisms have an extremely low probability. The approach to meeting this requirement is discussed at some length in the design characteristics and restraints section of Volume A. Ejecta from the propulsion systems of the VOYAGER Planetary Vehicle are one very obvious possibility of transporting viable terrestrial organisms to Mars. A comparison of the combustion temperatures, dwell time at high temperature, and the characteristics of the ejecta is given in Table 2.2-1. This data forms the basis for detailed analysis of particle trajectories and kill mechanisms to establish the probability of contamination of Mars by exhaust particles.

After some study of this question, it was concluded in Volume A that we can presently anticipate that it will be possible to avoid any requirement for sterilizing either the orbit insertion or orbit adjust propulsion systems of the VOYAGER Planetary Vehicle. Nevertheless, this conclusion is uncertain and subject to change. Consequently, the various propulsion system candidates were evaluated in order to assess the penalty to the VOYAGER program if a later requirement were imposed to sterilize the propulsion systems. Clearly, the propulsion system least affected by imposition of a requirement for sterilization would have some advantage in terms of minimizing program cost and schedule effects if further study does not

TABLE 2.2-1. PROPULSION SYSTEM EJECTA CHARACTERISTICS SUMMARY

System	Ejected Weight (lb)	Ejecta Utilization	Ejecta Composition	Temperature (Chamber / Exhaust)	Dwell Time	Exhaust Velocity
1. Orbit Injection Propulsion						
A. Solid	8400	90-second burning time resulting in Mars orbit	Combustion gas (Mol. Wt = 28.2 gm/mol) Al_2O_3 particles, insulation and exit cone liner char particles Freon 114B2 for TVC approx. 200 lb	5840°F/ 3830°F	0.01-0.10 sec.	9700 ft/sec (2.95 km/sec)
B. Liquid (Bi-propellant ablative chamber	8500	200-second burning time resulting in Mars orbit	Combustion gas (Mol. Wt = 20) Ablative chamber char particles carbonaceous particles of combustion	5500°F/ 3000°F	0.01 sec	9600 ft/sec (2.92 km/sec)
2. Midcourse/Orbit Adjust Propulsion						
A. Monopropellant	1900	MC No. 1 = 645 lb MC No. 2 = 105 MC No. 3 = 105 OA No. 1 50 OA No. 2 50 895	Decomposition Products (Mol. Wt = 14.1 gm/mol) 0.5% catalyst particles	1900°F/ 400°F	< millisec	7750 ft/sec
B. Bipropellant	1300	MC No. 1 = 510 lb MC No. 2 = 83 MC No. 3 = 83 OA No. 1 = 39 OA No. 2 = 39 754 lb Remainder utilized if required by mission	Combustion gas (Mol/ Wt = 20) Ablative coating particles, Carbonaceous particles	5500°F/ 3000°F	< millisec	9600 ft/sec (2.92 km/sec)

indicate that sterilization can be avoided. At this time, it is considered that the solid rocket motor with small monopropellant midcourse and orbit adjust system lead the liquid system with respect to planetary quarantine for the following reasons:

a. Because of the longer dwell time at the high temperatures in the solid rocket exhaust, a higher probability of destroying viable organisms during firing appears likely.

b. If complete orbit insertion propulsion sterilization should be required, the technology in this area is more advanced than the sterilization of large bipropellant propulsion systems. Investigation of sterilization of solid motors has been conducted on earlier programs, such as Surveyor. Extrapolation of these techniques to motors of the VOYAGER size does appear to be feasible, although it would be an expensive development program, and considerable difficulties should be anticipated. For the sterilization of liquid systems, the propellant must either be heated in their tanks, introducing pressure and reactivity hazards or else aseptic propellant transfer techniques must be developed. The problems of hardware compatibility with sterilization heating systems for the above chambers must also be defined and solved.

c. Although the present expectation is that sterilization of the midcourse and orbit adjust propulsion system will not be required either, the arguments that can be advanced against the requirement for sterilizing this system are not nearly as strong as the arguments available that sterilization of the main propulsion system will not be required. Hence, it is more likely that a requirement to sterilize the midcourse orbit adjust system will arise in the future. On this basis, the preferred system has an advantage over the LEM descent and Transtage configurations. There is clearly a distinct advantage in sterilizing only a small liquid propulsion system as opposed to sterilizing the large liquid system required for orbit insertion. One study has indicated possible sporicidal properties for monomethyl hydrazine, and another has indicated no degradation in performance after heating to 600°F. Aseptic propellant transfer and hardware sterilization methods have been investigated under the Surveyor program.

In conclusion, in terms of meeting the planetary quarantine restraint, it is considered that the solid propellant system with hydazine midcourse and orbit adjust engines would more easily satisfy planetary quarantine requirements if sterilization of the orbit adjust system is required.

2.2.2 Schedule Risk. With a fixed and unalterable launch period, it is mandatory that deliveries of flight-qualified articles be made on time. In this sense, the schedule makes no allowance for major unknowns that are not amenable to solution within the prescribed design and development time period.

The basis used to compare the systems regarding schedule risk was that of itemizing components which require additional development and to assess the risk involved.

2.2.2.1 Solid Propellant System

2.2.2.1.1 Solid Propellant Orbit Insertion Motor

a. Igniter - A new squib will be required to meet the VOYAGER specifications. This will necessitate minor modifications and requalification of the Safe and Arm mechanism. No new technology is involved, and the schedule risk is minimal.

b. Nozzle - The VOYAGER Orbit Insertion motor will operate for approximately 90 seconds. This represents an increase in burning time in excess of the Minuteman durations. The nozzle redesign for this increased burn time is straightforward, and solid propellant motors have been fired for longer duration. Proof of the nozzle design will result from full-scale motor firings. Sufficient time exists in the development program to allow for design modifications that may be necessary.

c. Propellant - The propellant modification required is a change in the oxidizer particle size blend to reduce the burning rate. Subscale motors have been fired with the particular blend selected, and no significant schedule risk is apparent.

   Tests are currently underway to evaluate the effect of prolonged vacuum storage on propellant properties. Present data on Minuteman vehicles stored for three years in silos indicated no degradation over this time period. No problems are currently seen in vacuum storage. Further, capping of the nozzle will maintain internal pressure in the engine to between 10^{-1} and 10^{-5} torr, values which permit ground demonstration of the storage effect.

d. Thrust Vector Control - Except for the cold-gas pressurization system, the TVC system is operational. The pressurization system presents no schedule risk, and the total TVC unit can be fully developed prior to any motor firings.

2.2.2.2 Midcourse Correction System. Components for the midcourse correction system, such as valves, regulators, etc., are common to monopropellant and bipropellant designs. As such, only the elements of the system not common to the two approaches are discussed below. Since existing components or modifications of such components are applicable to the design, the schedule risk concerned with valves and regulators is minimal. Stable operation of the total system is of greater significance. On the basis of past programs, no unusual problems that cannot be solved during the development program are foreseen.

2.2.2.2.1 Monopropellant Systems

a. Bladder life - Provided the tank temperatures are kept below 100°F, there does not appear to be any difficulty in storing hydrazine in butyl rubber bladders for protracted time periods. Exact limits should be explored to determine safety limits for thermal control design.

b. Catalyst life - The effect of vacuum storage on the life of the spontaneous Shell 405 catalyst needs additional investigation. No apparent problems are envisioned, but experimental verification is required.

2.2.2.2.2 Bipropellant Systems

a. Propellant Acquisition - A variety of potential solutions exists, such as aluminized teflon bladders, surface tension screens, rolling metal diaphragms, etc. However, on the basis of known data, none of these can be assessed as completely acceptable for VOYAGER applications without additional work. As a backup, the use of nitrogen settling jets can be considered, thus alleviating potential schedule risks.

b. Values and Seals for Nitrogen Tetroxide Lines - Because of the long-term contact between the oxidizer and the component parts in the plumbing line, careful attention must be given to the materials and use of high quality, high purity oxidizer. Based upon long-term storage experience with Titan II missile, the schedule risk is minimal.

2.2.2.3 LEMDE System

a. Pressurization System - A new high-pressure gas system will need to be developed. The design is straightforward and should present no difficulties which entail a schedule risk.

b. Propellant Acquisition and Slosh Control - The use of nitrogen settling jets as a backup should assure an acceptable solution to acquisition, but adequate slosh control of the propellant is a problem. Such attractive solutions as fine screens are not flight-demonstrated.

c. Titanium Stress Corrosion - Present indications are that the recent difficulties encountered with storage of nitrogen tetroxide in titanium tanks may be alleviated by careful control of the propellant. In any event, a change in tank material could eliminate these problems. A complete solution will probably be available well in advance of VOYAGER requirements.

2.2.2.4 Transtage System

a. Propellant line prevalves - A prevalve is presently under development by the Martin Company. No significant problems are envisioned.

b. Propellant Acquisition and Slosh Control - The comments about LEMDE on this subject apply here, except that the Transtage tank configuration lends itself somewhat better to the use of screens.

c. Vernier Engines - There are several flight qualified bipropellant engines in the 100-pound-thrust class which would be applicable. The major effort would be devoted to insuring stable operation of the system. Schedule risk should be minimal.

d. Pressurization System - The application of the blowdown system with makeup pressurization gas, while straightforward, will need considerable checkout. Since the thrust chambers are qualified over the range of chamber pressures expected, no major problem exists in this area.

2.2.3. Conclusion. The conclusion is that each of the propulsion systems studied require some modification to existing designs. Consequently, all must be considered to have potential development problems. It is expected that any of the candidate propulsion systems could be made available in time to meet the VOYAGER schedule with very little risk. Consequently, it is difficult to draw a very meaningful conclusion that shows any advantage in this area to a particular candidate. However, it can be argued that the modifications required to adapt the Minuteman engine to the VOYAGER Mission involves less development and schedule risk than satisfactory achievement of a thoroughly creditable means for propellant acquisition and slosh control in the large propellant tanks of LEMDE or Transtage when they are partially empty as in the VOYAGER Mission. Further, there is more background for designing an autopilot for a rigid body than for one involving propellant slosh modes. From this standpoint, it could be argued that there is more risk of schedule slippage in the design of an autopilot for the large liquid propellant systems.

2.3 Evaluation Against Competing Characteristics. After establishing that none of the competing propulsion candidates could be eliminated on the basis of failure to meet mission constraints, it was necessary to compare the several candidates in terms of their ability to satisfy the competing characteristics established by JPL. This subject is covered in this section.

2.3.1 Probability of Mission Success. This is the highest ranking priority for selection of Spacecraft System elements. This characteristic proved to be the dominant consideration leading to the selection of the preferred system. This subject will be treated by considering first the propulsion system reliability and backup modes available, and then by considering the other system implications of each propulsion system candidate.

2.3.1.1 Propulsion Reliability. In comparing propulsion system reliability, the use of numerical estimates of propulsion system reliability did not prove to be convincing. The first problem is that generic failure rates for components such as valves and regulators, are not especially meaningful when applied to components of such widely divergent characteristics as encountered when comparing the several candidate propulsion systems. Secondly, the vendor-provided reliability statements represent far different levels of design maturity; 88 Minuteman firings as compared with no flight data for the LEM descent propulsion system. Finally, all of the candidate propulsion systems would have to be modified for the VOYAGER application, with different effects upon the resulting reliability. Consequently, it is more valuable to confine discussion of the reliability aspects of various propulsion system candidates to a qualitative discussion of their merits.

It is possible to make several statements about the qualitative probability of mission success for each of the propulsion candidates which would meet with general agreement. In terms of the probability of successfully performing an orbit insertion maneuver, it was agreed that the modified Minuteman propulsion system rated highest. There are several reasons for this; the inherent reliability of the thrust producing process is highest for this system, and it is initiated by electro-explosive devices, rendering a very high probability of initiation. Shutdown of the basic thrust producing system is automatic upon the depletion of propellant; and control of the solid propellant engine thrust vector is through the use of several control valves instead of several control actuators, again, leading to the feeling of greater reliability.

In terms of the probability of successfully completing one or several midcourse maneuvers, the use of four monopropellant midcourse engines, with capability to perform with one pair not operating, is higher than the corresponding probability of success for the large bi-propellant liquid system because the midcourse maneuvers can be successfully completed in the preferred systems if one out of the four engines fail. This leads to a higher probability of success than reliance upon one out of one engine for the LEM descent candidates, or two out of two for the Transtage candidates.

However, the estimate that both orbit insertion and midcourse maneuver probabilities are individually higher for the preferred system does not lead to the conclusion that this is the most reliable candidate for the whole mission. This is because two series systems must both successfully operate in order to complete the mission, whereas, for the LEM descent system in particular, there is only one system which must operate to complete the mission. Hence, the argument again reduces to the probability of two systems out of two operating as opposed to one out of one. Now it is not obvious that the solid propellant candidate is the most reliable. In fact, it might be expected that the LEM descent system would show a higher estimate of reliability. For what it is worth, the numerical estimate of reliability prepared during the study verifies this expectation.

The qualitative judgement of Transtage reliability is more difficut. In this case, a vernier orbit correction and midcourse system is added. Moreover, the orbit insertion propulsion system requires that two thrust chambers out of two operate successfully to achieve the mission. A superficial examination of these statements might lead to the expectation that the Transtage is a less reliable way to accomplish the VOYAGER Mission. However, there are several mitigating factors that must be considered. First, the operation of the vernier engines is not essential to the mission success, because the execution of a midcourse maneuver can always be delayed until trajectory errors have propagated to the magnitude that the velocity change required is within the capability of the main propulsion system. Hence, failure of the midcourse propulsion system does not lead to mission failure, but only to greater operational problems, and perhaps reduced trajectory accuracy. Moreover, it is possible to utilize these vernier engines in a manner that permits pair-out operation. Hence, the midcourse capability of the Transtage candidates is backed up twice, compared to once for the solid propellant system and not at all for the LEM descent systems.

In terms of the orbit insertion maneuver, the Transtage requires that two out of two thrust chambers operate successfully. Even here, however, there are mitigating factors. The candidate systems considered utilize the large ullage volume as explained before to simplify the propellant pressurization system. Specifically, the Transtage candidates do not require the use of a high pressure gas regulator. Since this is one of the most unreliable portions of a liquid propulsion system, its removal is a distinct reliability advantage.

Another interesting comparison of the propulsion system candidates can be made on the basis of the original mission for which each system was designed. The Minuteman solid propellant engine was designed for reliable response after a long unattended storage. Even though the storage environment is under one g and earth atmosphere, this original design intent must count in its favor in a study of the applicability to the VOYAGER Mission. However, this propulsion system was not designed with man-rating capability in mind, which is a minus factor in comparison with the LEM descent systems. The LEM descent propulsion system was designed to be man-rated, implying a very high reliability goal. However, long space storage life is not a design criteria for the LEM descent system. The complications of extending the space life of the LEM descent system from several days in space to many months in space is not expected to be a very large problem, but should be remembered in comparing propulsion system reliability. The Transtage was designed for only a few hours of life in space, and is not designed as a man-rated system. These factors must count against it. However, partially offsetting these drawbacks is the fact that much of the technology embodied in the Transtage is directly derived from the Titan II missile technology. For example, materials compatibility, and leakage through pressurization valves, are areas in which the technology of the Titan missile has been directly transferred to the Transtage. This counts in favor of the Transtage reliability because Titan, like Minuteman, is designed for instant response after long unattended earth storage.

It should be recognized at this point that the reliability discussion thus far has assumed equal experience for all of the candidate propulsion systems. In fact, this will not be the case. Each of the propulsion system candidates will require appreciable modification for the VOYAGER program. It is worthwhile to examine the nature of modifications required for each system and qualitatively judge the effects they will have upon reliability of the candidate systems. This is shown in Table 2.3-1, which indicates the major modifications required, and indicates the expected reliability effect.

After the modifications discussed in Table 2.3-1 have been completed, a test program will be required to verify the adequacy of the modified design. If the testing program is extensive enough to bring all of the propulsion systems to the same design maturity, the qualitative

TABLE 2.3-1. MODIFICATIONS REQUIRED FOR CANDIDATE PROPULSION SYSTEMS

System	Major Modification	Reliability Implications
TRANSTAGE	a. Vernier thrust chambers b. Propellant acquisition screens c. Propellant feed line pre-valves d. Blowdown pressurization system.	a. Four thrust chambers with pair-out capability increases reliability for MC/OA maneuvers. b. Unproven in flight, but if proven practical would improve reliability through simplicity. c. A fix for inadequate main shutoff valves — it would degrade reliability since a single pre-valve failure, open or close, could result in mission failure. d. Eliminates gas regulator which is one of the highest risk components. Lower pressure gas storage also contributes slightly to overall reliability.
LEMDE 1. "As-is" Configuration	a. Return to original high pressure helium storage instead of cryogenic helium storage. b. New gimbal actuators for higher response rates needed by VOYAGER. c. Propellant acquisition system required. d. Dual squib shutoff valves (NC) in main propellant lines.	a. Subject to all reliability problems associated with high pressure helium storage and regulated helium systems. b. Standard design approach should involve no reliability problems. c. Reliability dependent on selected system. Use of screens, if proven, give a system that is attractive through its simplicity. d. Improves reliability through positive sealing of propellant during launch phases.
2. LEMDE Modified	a. Same as (1), above, but with new structure and four shortened tanks.	a. Structure and tank modifications should have no effect on reliability.
SOLID MOTORS Modified Minuteman (Aerojet)	a. Decreased chamber barrel length b. Decreased burning rate c. Increased burning time d. Nozzle throat diameter increased e. New squibs in the igniter f. Cold gas pressurization for TVC system g. Power for servo control unit derived from bus power instead of self-contained battery h. Smaller tank for Freon	a. None b. Ballistic evaluation necessary - established technique. c,d. Aft case insulation and nozzle require requalification-burn time well within present state-of-the-art e. Increased reliability f. Increased reliability over warm gas pressurization system g. Increased reliability h. None

conclusions discussed at the beginning of this section would remain valid. However, in examining the nature of the changes required, it is expected that the changes required to the Minuteman system will introduce less reliability concern than changes to the liquid systems. This, taken with the already greater design maturity of the Minuteman stage, indicates that for programs of roughly equivalent cost, the modified Minuteman propulsion system might be expected to have the greatest post modification reliability of any of the candidates. As indicated, this conclusion could be altered if a sufficiently extensive testing program were undertaken on the liquid system. Although this is unlikely, because of cost, the final conclusion is that the available reliability from various propulsion system candidates is perhaps more of a cost related factor than an inherent reliability problem.

The other point which should be made in this connection is that the long life storage and zero g acquisition of hydrazine monopropellant is already flight demonstrated to a much higher degree than is the case of the liquid bipropellant system candidates. In short, although the inherent reliability potential of the LEM descent design is probably the highest, the actual reliability achievement during the VOYAGER development program probably favors the Minuteman system to a small degree.

Each of the three propulsion systems studied; LEMDE, Transtage, and Solid Retro/Monopropellant Midcourse can be brought to an acceptable level of reliability through proper redesign and testing procedures. The Solid Retro/Monopropellant Midcourse system is the preferred choice however, because of maximum application of existing design, minimum number of functions to initiate and terminate firings, and minimal development problems.

2.3.1.2 Other System Reliability Effects. In addition to the propulsion system probability of success considerations, the other most significant system effects involved in the selection of a propulsion system candidate are the effect on autopilot design, attitude control of the Planetary Vehicle, thermal effects, and reliability considerations inherent in the configuration differences dictated by various propulsion system candidates.

2.3.1.2.1 Autopilot. - The differences in autopilot configuration are the result of differences in the propulsion systems on which they are based. Table 2.3-2 compares these configurations based upon the techniques used for obtaining control torques.

TABLE 2.3-2. AUTOPILOT CONFIGURATIONS COMPARED ON THE BASIS OF TORQUE CONTROL MEANS

	Correction Maneuvers		Orbit Insertion	
	Pitch/Yaw Torques	Roll Torques	Pitch/Yaw Torques	Roll Torques
Preferred Design	Monopropellant Engine Vanes	Monopropellant Engine Vanes	Secondary injection of Freon	ACS Roll Jets
Transtage Designs	Four gimballed Engines	Differential Operation of Small Engines	Two gimballed Engines	Differential operation of gimbals of Main Engine
LEM Descent Engine Design	Gimballed Main Engine (low Thrust)	ACS Roll Jets	Gimballed Main Engine (at high thrust)	ACS Roll Jets

Despite the differences in characteristics depicted in Table 2.3-2, autopilots can be designed well within the requirements of the overall guidance system for each of the vehicle configurations. The autopilot for each configuration has the same sensing requirements (i.e., attitude information from gyros). Each vehicle configuration would have a high and low thrust level operation; and in no case is simultaneous operation of high and low thrust level systems planned as a primary approach. Finally, in no case are autopilot parameter changes (e.g., gain changes) required for operations at a given thrust level.

The preferred design requires no autopilot switching operation for accommodation of the selected thrust level. Transtage and LEMDE designs do require such switching operations. Analysis of the Transtage and LEMDE configurations is more complex than the preferred design because they must include such dynamic coupling effects as engine inertial coupling and possible motion of unrestrained propellants. Because of nonsymmetrical tank shapes, the Transtage design experiences a larger lateral shift of cg than do either of the other two. Due to shut-down variation of two engines, the response of the Transtage designs may exhibit larger nonsymmetrical thrust termination torques greater than either of the other two designs. These could lead to terminal roll rates up to 1.7 degrees/second and yaw rates up to 5.5 degrees/second.

A significant autopilot design consideration that differs from the preferred design relates to the effect of propellant sloshing on the autopilot characteristics during engine burn for the large liquid propellant system candidates. After some study of this problem, it was concluded that this was not a major reliability concern for the large liquid propellant systems. The control torques available to maintain the thrust vector in the desired direction in inertial space are large compared with the slosh effects of the unrestrained propellants. Consequently, this problem is qualitatively like the autopilot design for a large Launch Vehicle, such as Atlas or Titan. In this sense, the autopilot solution is state-of-the-art, and not a cause for significant concern about reliability.

One other significant autopilot design difference should be noted. The autopilot for the LEM descent propulsion system has an advantage in that it must provide fewer control outputs to the propulsion system. It is required to provide only one pair of outputs to control pitch and yaw of a single engine. The preferred design must add to this 3-axis control for the mid-course maneuver engines; and Titan requires even more outputs since roll control during main engine firing is accomplished by differential operation of the main engine gimbals. Consequently, the LEM descent autopilot would be somewhat less complex; thus, it might be argued that it could be made slightly more reliable.

The conclusion drawn as a result of this study is that there is no significant reliability difference among the various propulsion system candidates in terms of the effect upon autopilot design.

2.3.1.2.2 Attitude Control. - In the area of attitude control of the Planetary Vehicle, there is a large difference between the solid and liquid propellant orbit injection propulsion systems. If the large liquid engines are selected for orbit insertion, a very significant fraction of the Planetary Vehicle mass is liquid, with some ability to create disturbance torques through sloshing modes. The exact magnitude of this problem depends upon the measures taken to assure propellant acquisition for engine starting and the related effect upon propellant sloshing. Both the LEM descent and Transtage propulsion systems have propellant tankage volume substantially in excess of that required for the VOYAGER Mission. Consequently, both of these propulsion systems when applied to VOYAGER are being flown with tanks that have substantial ullage volume.

There are conceptually two approaches to provide propellant acquisition for the liquid propellant systems. The first of these is the use of propellant settling thrustors, to assure propellant presence at the tank outlets. This method has the greatest assurance of success, and is flight demonstrated. (It should be noted that the use of this propellant acquisition technique is not clearly the most reliable, despite the statement that it is the most assured of working. In the case of propellant settling thrustors, an additional system, which must work, is added in series to the other mission sequence events. This has an adverse effect upon reliability.) If propellant settling thrustors are the selected means of propellant acquisition, the propellants may be left unrestrained during Planetary Vehicle cruise. This would pose a major problem for attitude control of the Spacecraft, because of both the large propellant slosh disturbances, and the uncertain position of the Spacecraft center of mass. This problem could be mitigated, but not eliminated, by the addition of extensive baffling within the propulsion system tanks.

The second conceptual family of propellant acquisition techniques relies upon equipment within the tank to control the interface between propellant and pressurizing gas. Propellant control screens have been advocated for this purpose, and there is considerable laboratory test experience to justify the belief that this technique would be successful. In addition, there is some flight experience on the Transtage program which re-enforces this expectation.

Nevertheless, this technique has not been flight proven for the durations involved in a VOYAGER Mission. Furthermore, because of the repeated firing of the liquid propellant systems, the level of propellants in the tank varies throughout the mission in a number of discrete but somewhat unpredictable steps. This will require propellant control screens to be installed at several positions within the tanks if propellant acquisition is to be assured for each firing, and excessive sloshing of the unconsumed propellant minimized.

The other family of in-tank propellant acquisition devices is the use of bladders or diaphragms. Many configurations have been considered, including elastomeric bladders, rolled metal diaphragms, and thin rupture diaphragms. Each of these propellant acquisition techniques provide restraint, in varying degrees, to the unconsumed propellant. Unfortunately, the most nearly qualified of these techniques offers the least propellant restraint. Consequently, in the final tradeoff between reliable propellant acquisition and propellant slosh control, the optimum balance probably lies near a point where the propellant is inadequately restrained during interplanetary cruise, resulting in a large problem for the attitude control system. In conclusion, it was felt that the use of a large liquid propellant orbit injection system would be a serious reliability penalty to the attitude control system. This was one of the major factors involved in the selection of a solid propellant system. It should be noted, however, that the adoption of the preferred system does not entirely eliminate this propellant slosh problem, insofar as the monopropellant midcourse maneuver fuel also poses a fuel sloshing problem to the attitude control system. However, in this case, it can be analytically demonstrated that the fraction of Planetary Vehicle mass which is not restrained is sufficiently small that effective attitude control can be maintained even with unrestrained propellants.

It should be pointed out why the slosh problem did not preclude selection of a liquid propellant engine for the Task A report. There are several significant differences between the Task B design and the Task A results. First, the amount of liquid was a much smaller percent of the total vehicle weight. The liquid orbit insertion engine of the Task A design consumed all propellants at the orbit insertion burn. Further, the tankage design for the Task A study was designed specifically for the planned VOYAGER Mission. In other words, the propellant tanks were full throughout the Planetary Vehicle cruise phase thus minimizing slosh during the cruise portion of the mission. After the orbit insertion burn the tanks were nearly empty, except for unavoidable outage. This meant that the mass of unrestrained propellant, after orbit insertion, was a sufficiently small portion of the Spacecraft mass to pose no control problem. The sloshing of midcourse propellants of the Task A study was a problem similar in proportion to that of Task B; that is, the unrestrained mass was a small fraction of the total vehicle mass.

2.3.1.2.3 Thermal - The heat radiated from the engine exhaust plume onto the solar array during retro fire for orbit insertion causes a temperature rise of the array. The worst case situation is with the solid retro propulsion system. This effect requires additional investigation, although it is presently believed that the problem is not one of very large magnitude. This topic is covered in more detail in Volume A.

At the completion of retrofire, the solid motor case will be at a temperature of 790°R. The effects of this condition on the system were the subject of a worst case thermal analysis which revealed that the temperature of bay 12 would rise to a maximum transient value of 586°R, which would cause the traveling wave tube of the telemetry transmitter to approach, but not exceed, its maximum rated operated temperature of 175°F. No other temperature rise of significant effect on the system was determined and it was, therefore, concluded that the solid rocket motor case temperature does not represent a system reliability problem.

2.3.1.2.4 Configuration - The inherent differences among the candidate propulsion systems dictated many differences among the optimum Spacecraft configurations for each candidate. These differences were analyzed from the standpoint of their effect upon the reliability of the overall system. Only two of the configuration effects examined appeared to have any significant bearing upon overall mission reliability.

The most obvious reliability effect for any system is the fact that the solar power panels must be deployed on the unmodified LEM descent configuration. This is in addition to the deployment of antennas and other Spacecraft elements. Failure of the solar array to deploy, results in mission failure, so the addition of this requirement to the configuration is a significantly adverse effect upon mission reliability.

One other reliability effect, of lesser magnitude, exists for the Transtage configuration. For the configurations which were entered into the final tradeoff analysis, the Transtage configurations extend considerably below the separation plane. In other words, the Spacecraft must "fly out of a hole" at Launch Vehicle separation. During this period, there is relatively little clearance between the exhaust cone of the thrust chambers and the adapter ring. This implies some danger of a damaging collision between the Spacecraft and adapter at separation. This danger can be eliminated by the addition of guide rails to the configuration, but this also implies a decrease in separation reliability.

2.3.1.3 Probability of Success Conclusions. The result of this study of the probability of mission success results in two conclusions.

a. Any of the proposed propulsion system candidates can be made to yield a very high probability of mission success for the overall system.

b. Considered on a total system basis, the preferred configuration has the best overall probability of successfully accomplishing the VOYAGER Mission. This was the dominant factor in the selection of the preferred system.

2.3.2 Performance of Mission Objectives. In considering the ability of each of the five propulsion candidates to perform the VOYAGER Mission, two significant factors emerged: Less accurate performance of midcourse maneuvers by the LEM descent system, and greater velocity flexibility for liquid orbit insertion engines as compared with the selected design. These will be discussed in turn. The LEMDE propulsion system in the "as-is" configuration is designed for multistart operation and to provide thrust modulation capability from 10,500 pounds to 1050 pounds. Although there is a lack of precise information on the LEMDE engine, it can be assumed that its minimum impulse bit capabilities and accuracy of cut-off is comparable to that of other engines in the same thrust category. Based on characteristics of similar thrust chambers, it should be possible for the LEMDE engine to obtain impulse bits of 500 pound-seconds when operating at the minimum thrust level. A 500-pound-second impulse would provide about 0.25 meter/second ΔV for MC and one meter/second ΔV for OA functions. This satisfies the system requirement of one meter/second ΔV for midcourse maneuvers, but is poorer than the system goal of 0.1 meter/second met by Transtage and the preferred design. However, the mission can be satisfactorily completed with LEMDE performance.

There are two aspects of the orbit insertion maneuver with a fixed impulse solid propellant system worthy of comment. Where the total retro-impulse is fixed, the orbit insertion velocity capability varies with the amount of propellant used in midcourse maneuvers. There are two ways to overcome this handicap. First, guidance and trajectory studies have indicated the variation of insertion velocity which will result from unpredictable midcourse propellant usage can readily be accommodated by adjusting the aiming point in the R-T plane slightly at the time the last trajectory correction is made. This appears to be the simplest and most expedient way to accommodate varying midcourse propellant usage. However, it should be noted that the orbit insertion solid propellant system has been sized to achieve the desired insertion velocity with all midcourse propellants used. If they have not been used, the velocity available will be less. However, the deficiency will always be less than the amount of velocity capability remaining in the midcourse propellant tanks, so the deficiency can always be made up by the firing of the midcourse engines, either at the time of orbit insertion or at a subsequent orbit injection maneuver.

The second aspect of fixed impulse orbit insertion is that this maneuver may be required after the capsule has been jetisoned. The solid propellant system, having no shut down provisions, will impart excess orbit injection velocity to the Spacecraft. Again, however, our trajectory studies have indicated that it is always possible to inject the Spacecraft into a useful orbit with a fixed velocity solid propellant system.

The conclusion is that all of the candidate systems are capable of performing the mission objectives within the design constraints. The differences among the various propulsion system candidates in their ability to satisfy the VOYAGER Mission requirements was not a significant consideration in the selection of the preferred design.

2.3.3 Cost Savings Comparisons. In the selection of the preferred propulsion system candidate, careful attention was given to the third priority JPL competing characteristic, cost saving.

The cost data presented herein are for purposes of comparison and are based upon budgetary costs submitted by the propulsion contractors who participated in this study with the General Electric Company. Information on costs of the LEMDE propulsion module other than "as-is" delivery costs were unavailable. However, considering the relative sizes, system complexity, and required modifications, it is reasonable to assume that the LEMDE costs are equal to those quoted for the Transtage.

Table 2.3-3 is a tabulation of propulsion system costs for the propulsion units under study. From Table 2.3-3, it may be seen that design, development, and T/A costs are nearly equal for all systems considered. Total costs through 1971 indicate that the solid/bipropellant combination is approximately 7% higher than the solid/monopropellant configuration.

The Transtage or LEMDE propulsion modules are about 20% more costly compared with the solid/monopropellant system. The table also indicates that the same relative ranking exists through 1977. Shown in Table 2.3-4 are budgetary estimates for unit costs of each propulsion system for the 1971 flight articles.

TABLE 2.3-3. PROPULSION SYSTEM COST COMPARISON

Propulsion System	Estimated Cost, Millions of Dollars							
	Design Development T/A			Total Cost Through 1971 Delivery			Total Cost Through 1977 Minimum Modification	Total Cost Through 1977 Maximum Modification
Propulsion System	Maximum	Minimum	Mean	Maximum	Minimum	Mean		
Solid Retro Unit	9.3	6.7	8.0	11.1	7.1	9.1	-	-
Midcourse/Orbit Adjust								
Monopropellant	6.9	4.6	5.8	9.3	5.9	7.6	-	-
Bipropellant	8.5	4.4	6.5	10.7	6.8	8.8	-	-
Solid + Monopropellant	16.2	11.3	13.8	20.4	13.0	16.7	23(1)	33(2) - 40(3)
Solid + Bipropellant	17.8	11.1	14.5	21.8	13.9	17.9	26(1)	36(2)
Transtage/LEMDE	-	-	13.0	-	-	20.0	30(1)	45(4)

Notes: (1) No basic system changes
(2) Change solid propellant to beryllium
(3) Change solid propellant to beryllium, Change monopropellant to bipropellant
(4) Change to high energy propellants

TABLE 2.3-4. PROPULSION SYSTEM UNIT COSTS (1971)

Propulsion System	Unit Cost (1971) Thousands of Dollars		
	Maximum	Minimum	Mean
Solid Retro Unit	417	130	274
Midcourse/Orbit Adjust			
Monopropellant	300	272	286
Bipropellant	410	270	340
Solid + Monopropellant	717	402	560
Solid + Bipropellant	827	400	614
Transtage			1320
LEMDE			3310

Two significant factors emerged from this comparison. The first point to be noted is that the cost difference between the various propulsion system candidates is only a few million dollars. Compared with the overall development cost for the VOYAGER Planetary Vehicle, the cost differences between the various systems are too small to have a significant bearing on the selection of the desired propulsion system.

The second significant factor whick emerged from the study was that contrary to initial expectations, the adaptation of the existing Transtage or LEM descent propulsion system to the VOYAGER Mission did not result in a cost saving in comparison with the use of a solid propellant system with a new midcourse and orbit adjust system. On the contrary, modification of the existing liquid propulsion systems was estimated to be more costly than use of the preferred system.

In conclusion, cost factors favor the selection of the preferred system, although the size of the savings and the significance of this factor did not weigh heavily in the final decision.

2.3.4 Contribution to Subsequent Missions. Consideration of this subject was confined to evaluating the orbit insertion velocity that would be available for Mars missions in the 1975-77 time period. Velocity data was calculated, and is shown in Table 2.3-5. This data is predicated upon a maximum total propulsion system weight of 15,000 pounds and a 13,500-pound payload (Bus plus Capsule) and the same velocity requirements for midcourse corrections, and orbit adjust.

TABLE 2.3-5. 1975-77 PROPULSION SYSTEM PERFORMANCE

Propulsion System	Orbit Insertion ΔV, (km/sec)
Solid (Aluminum propellant)	
Monopropellant Midcourse (Preferred 1971 System)	1.42
Bipropellant Midcourse	1.52
Solid (Beryllium Propellant)	
Monopropellant Midcourse	1.49
Bipropellant Midcourse	1.59
LEMDE	
"As-Is"	1.33
Modified Four Tanks	1.52
Transtage	
U-E3	1.28
MC-3 (Shortened Tanks)	1.30

Several conclusions can be drawn from the results in Table 2.3-5. First, all of the propulsion system candidates would provide sufficient orbit insertion velocity to achieve useful Mars orbits for the 1975 and 1977 Missions. Second, Transtage configurations are significantly poorer than the best of the other configurations in this regard. Third, the preferred 1971 system provides an orbit insertion velocity competitive with the LEM descent system, and by expanding the effort for a development program to change to beryllium propellant, the solid propellant orbit insertion capability is only slightly smaller than the best

LEM descent system. The fourth conclusion is that if the maximum orbit insertion velocity is desired for the later missions, this can be provided by the beryllium propellant solid rocket system by paying for the additional development effort involved in providing a bipropellant midcourse engine for the subsequent missions.

The significance of an additional 100 meters/second or so of orbit insertion velocity must await further study of the subsequent missions. Consequently, it is difficult to assign any quantitative significance to the data of Table 2.3-5.

In summary, the preferred system selection is competitive with the liquid propulsion candidates in providing contributions to subsequent missions, but no great significance was attached to this characteristic in making the final selection.

2.3.5 Additional 1971 Mission Capability. The propulsion system candidates entered into the final tradeoff all provide the required 1971 Mission capability, and none of them were endowed with any additional capability beyond the goals set in the JPL mission description. Consequently, they must all be evaluated as equal in this regard. It should be noted, however, that any of the systems could provide additional orbit insertion velocity or additional payload if this were required.

2.3.6 Summary. Table 2.3-6 summarizes the discussion of the preceding sections. The factor most strongly favoring the use of a solid propellant orbit injection system was the expectation that this would provide the greatest overall spacecraft probability of success.

This conclusion, in turn, reflects the severity of the propellant slosh problem, and the high degree of design maturity in the selected solid propellant engine. The other factors favoring the selection of the preferred system were easiest satisfaction of the planetary quarantine constraint, and somewhat lower cost for this system.

TABLE 2.3-6. PROPULSION SELECTION CONCLUSIONS SUMMARY

	Minuteman Solid Plus Mono MC/OA	LEMDE	Transtage
Constraints			
• Planetary Quarantine	Best • Most technology • Small MC/OA engines • Most likely sterile exhaust	Acceptable	Acceptable
• Minimum Schedule Risk	Acceptable	Acceptable	Acceptable
Competing Characteristics			
• Mission Success	Best Overall Rating • Mature design • No propellant slosh	Best propulsion system reliability • Single engine design	Acceptable
• Mission Performance	Meets Goals	Meets Requirements • 0.25 meters/sec min ΔV • More Impulse Flexibility	Meets Goals • More Impulse Flexibility
• Cost	Lowest	Most Unknown	Acceptable
• Contribution to Later Missions	Acceptable	Acceptable	Acceptable
• Additional 1971 Capability	None	None	None

3.0 CONFIGURATION. This section describes the Spacecraft configuration studies that were conducted in arriving at the recommended design. Configuration layouts were made for each propulsion system considered, with several alternative configurations being considered for each case. These configurations were then compared on the basis of such factors as structural simplicity, overall reliability, vehicle length, weight, inertias, and ease of fabrication and assembly.

While some of the configurations have obvious advantages relative to others, no demanding reasons were found for choosing one propulsion approach over the others from a configuration standpoint. That is, configurations can be derived for each propulsion approach that will adequately satisfy all requirements.

General design criteria that were established for all configurations are as follows (not in order of priority):

a. Must fit within specified Spacecraft envelope.

b. Minimize overall Spacecraft length.

c. Near uniform loads at the interfaces with the shroud and the capsule.

d. Capsule location on shady side in cruise orientation.

e. A single planetary scan platform, mounted on the edge of the solar array, with two Canopus sensors to provide full-planet viewing capability over the mission lifetime.

f. Propulsion system on the longitudinal axis with the nozzle pointed at the Sun in cruise orientation.

g. Electronic equipment mounted in a torus with 16 sides.

h. Fixed solar array with 16 structural panels.

i. Deployable, steerable 7 1/2-foot high-gain antenna.

j. Minimize Spacecraft weight and inertias.

k. The Spacecraft center of mass should nominally be on the longitudinal axis and sufficiently forward of the engine gimbal point to ensure proper autopilot operation.

l. A high degree of modularity is desired to provide ease of fabrication, assembly, test, and repair.

The degree to which these criteria were met formed the basis for selecting the most promising configuration associated with each propulsion approach.

3.1 Solid Retropropulsion System General Description. The solid propellant configurations studied reflect primarily the attempts to integrate the modified Minuteman engine into a well-balanced VOYAGER Spacecraft design. Some initial efforts were expended on an ovaloid shape submerged nozzle engine of new design, described in Section 4.0. This engine lends itself to development of a compact spacecraft design in which the engine mount diameter would be compatible with the basic structural load path. However, the development status of this type of engine was not considered sufficient to warrant selection of a new engine for the 1971 design, and these designs were not seriously considered for the selection of a preferred system.

The existing Minuteman engine lends itself to uncomplicated modification for orbit insertion of the various future VOYAGER vehicles, by simply adjusting the cylindrical length of its case. The 52-inch diameter leaves adequate volume remaining for the separate midcourse and orbit adjust system within the body of the model Spacecraft discussed in Section 2.0. A schematic showing propulsion system components is presented in Figure 3.1-1.

3.1.1 Specific Requirements Which are Pertinent Constraints to the Minuteman System

a. Requirement for provision of a separate modularized midcourse and orbit adjust system sized for the 1975/77 Mission.

b. Use of a liquid injection TVC system rather than engine gimbaling.

c. Exhaust plume with high radiant heat transfer to the spacecraft.

d. Requirement for installation of the main engine late in the flow cycle.

e. Higher thrust levels associated with a solid propellant engine.

3.1.2 Resulting Configurations. Major configuration influencing alternatives were the location of the solid engine with respect to the Flight Capsule (i.e., whether the room for growth of the 1975 engine would be forward or aft), the high-gain antenna stowage location, and the shape of the Planetary Vehicle adapter (inverted cone versus upright cone, and cone angle). The more promising configurations developed from the study are presented with brief descriptions in Figure 3.1-2.

The selection process which led to choosing of Configuration G as the most promising solid propellant system is as follows:

a. Configurations C and E were considered less attractive from a configuration standpoint due to interference with the antenna pattern by the main retro-nozzle. Configuration C has a potential problem of blockage of fixed solar array surface area if the antenna fails to deploy, and E removes fixed solar array area to provide room for the antenna.

b. Configuration F was ruled out as a contender due to the lengthening of the vehicle and associated added weight and lower stiffness parameters, whereas the only advantage was the possible use of the high-gain antenna for a period of time after encounter even though deployment fails.

c. Configurations A and B are essentially the same except for the location of the solid motor for 1975 growth, the method of supporting the motor, and possible access to the interior from the aft end. The decision was made to move the engine to the forward end and provide for 1975 growth to the rear in the interests of minimizing inertia and this consequently removed Configuration B.

d. This then left configurations A and G as the final contenders with the only basic difference between them being the conical adapter aft of the solar array versus the

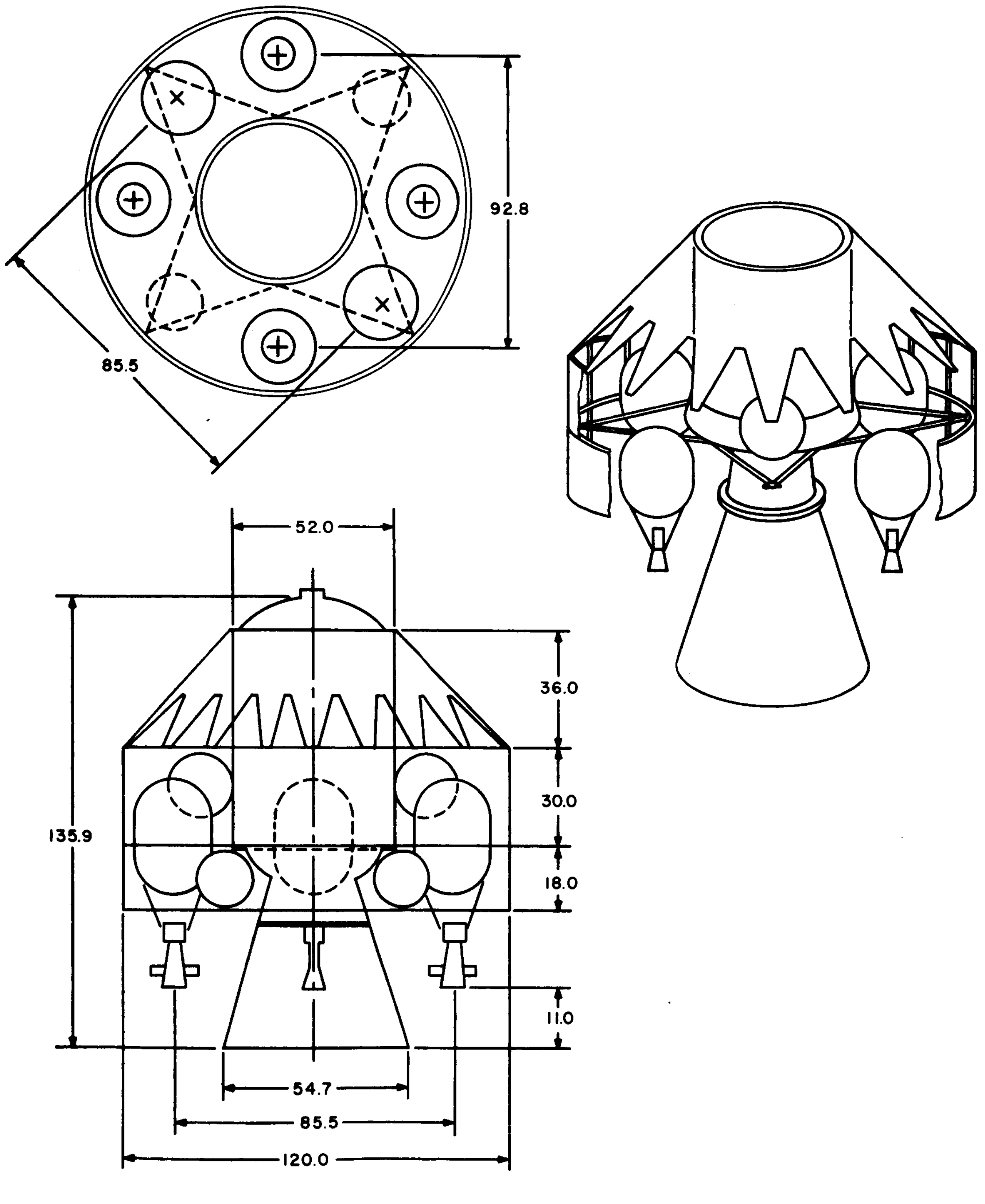


Figure 3.1-1. Propulsion System Components

truss adapter forward of the solar array. The advantages of the Configuration G which led to its selection as the preferred solid propellant configuration are:

1. Much shorter overall Spacecraft and subsequent minimization of shroud length.

2. Inherent structural stiffness due to shortening of load path.

3. Better control of the solar array envelope with respect to the Space Vehicle dynamic envelope, better support for all solar array mounted components during the boost environment, and greater flexibility for solar array surface areas.

4. Less separation problems due to no requirement for flying out-of-the-hole. In Configuration A, the Midcourse and Orbit Adjust engines have relatively tight clearance with respect to a 10-foot diameter adapter upper ring.

5. Less possibility of damaging the solar array surface during separation.

e. The disadvantages of Configuration G with respect to A are as follows:

1. The truss adapter really becomes part of the Flight Spacecraft and, therefore, becomes Cruise and Mars orbital weight in turn requiring increased propellant weight. However, the net weight to the Space Vehicle system is less when the reduction of shroud and adapter weight are taken into account.

2. The more concentrated loads at the shroud interface, deviates slightly from the suggested criteria.

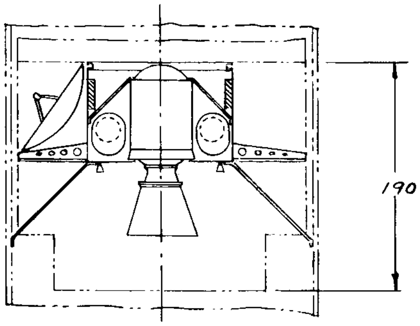
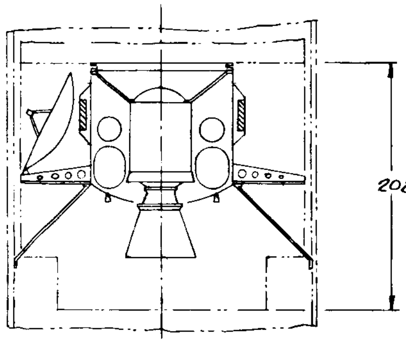
3. The solar array panels are required to carry shear loads, but this is basically a more efficient design since a multifunction structure is employed. The inherent disadvantage is in the case where shear loads may occur during ground handling operations when the spacecraft is supported at the 20-foot diameter. However, the OSE plans call for support of this system at the 10-foot diameter prior to solid motor installation, and for all other cases separate OSE panels can be installed as necessary.

3.1.3 Description of Selected Configuration. The basic structure, described in Volume A, is a 120-inch diameter cylindrical shell with 16 longerons and rings at the two manufacturing joints; these joints divide the structure into the three basic modules; the upper electronic equipment module, the midcourse and orbit adjust system module, and the solar array module. The Spacecraft is supported at the 240-inch launch vehicle interface diameter, and boost loads are transmitted via support tubes connecting the attachment fittings to the joint between the midcourse module and the electronic equipment module. The solid retropropulsion engine is supported at its head end by a conical structure which assembles immediately adjacent to the aforementioned support tube attachment fittings at the joint between the midcourse module and the electronic equipment module. Lateral support is provided for the engine at its aft skirt, by a system of struts, connecting to a ring of the 120-inch diameter basic cylindrical structure. The electronic equipment is housed in 16 integrated assemblies in the toroidal upper structure. The engine support conical structure also supports the main harness which is of significant weight. This structure is, therefore, assembled to the toroidal electronic equipment structure forming a modular electronic unit.

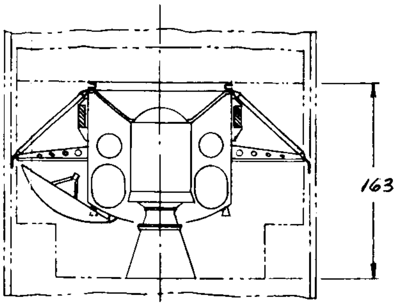
PROPULSION SELECTION MATRIX

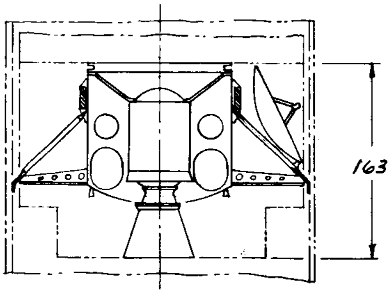
MODIFIED MINUTEMAN CONFIGURATIONS

DESIGN FEATURES	
TOTAL EARTH TAKE-OFF WEIGHT	
ADAPTER - S/C TO BOOSTER - WEIGHT	
TOTAL CRUISE WEIGHT	
BURN-OUT WEIGHT	
PROPULSION SYS. TOTAL WT.- EMPTY	
PROPULSION SYS. TOTAL WT.-LOADED	
BUS WEIGHT	
ENVIRONMENTAL SHIELD WEIGHT	
SOLAR ARRAY AREA - FIXED	
SOLAR ARRAY AREA-DEPLOYED	
RECOMMENDED SEPARATION SYSTEM	
PLUME IMPINGEMENT	
NATURAL FREQUENCY - FIRST MODE	LONGITUDINAL
	LATERAL
	TORSIONAL

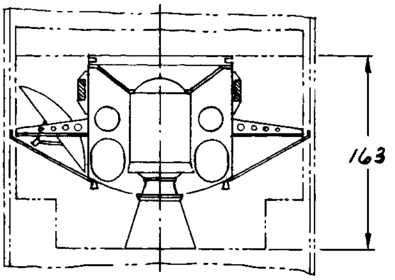
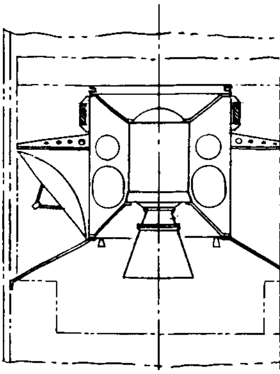
 CONFIG. "A"	 CONFIG. "B"
BASIC STRUCTURE :- 120" DIAMETER CYLINDRICAL SHELL. 16 LONGERONS. RINGS AT 2 MFG. JOINTS. ELECTRONIC EQUIPMENT IN 16 INTEGRATED ASSEMBLIES IN TOROIDAL STRUCTURE. RETRO-ENGINE SUPPORTED AT HEAD END BY UPRIGHT CONE STRUCTURE. MONO-PROPELLANT MIDCOURSE SYSTEM IN CENTER MODULE. SOLAR ARRAY IN 15 PANELS. MODULAR STRUCTURE HOUSES ATTITUDE CONTROL SYSTEM. 45° HONEYCOMB SHELL CONICAL ADAPTER SUPPORTS SPACECRAFT AT 120" DIAMETER.	SIMILAR TO "A" EXCEPT RETRO-ENGINE IS SUPPORTED AT HEAD END BY AN INVERTED CONE STRUCTURE, LOWERING THE ENGINE AND PROVIDING MORE SPACE FOR TANKAGE SUPPORT STRUCTURE. MICROMETEORITE BARRIER IS DISHED.
17,503 #	17,577 #
336 #	336 #
17,207 #	17,281 #
4,232 #	4,255 #
1,525 #	1,525 #
11,500 #	11,551 #
2,637 #	2,660 #
70 #	70 #
225 ft^2	230 ft^2
NONE	NONE
MILD DETONATING FUSE	MILD DETONATING FUSE
NOMINAL	NOMINAL
———	15.9 C.P.S.
———	7.0 C.P.S.
———	18.0 C.P.S.

✓





CONFIG. "C"	CONFIG. "D"
INTERIOR IDENTICAL TO "B"; SPACECRAFT IS SUPPORTED BY INTEGRAL TUBULAR STRUCTURE, ELIMINATING ADAPTER. SOLAR ARRAY ELEVATION IS AT CENTER OF MODULE - TIES IN TO TUBULAR STRUCTURE.	INTERIOR IDENTICAL TO "B"; EXTERIOR SIMILAR TO "B" EXCEPT ADAPTER ELIMINATED AND TUBULAR TRUSS STRUCTURE ADDED TO SUPPORT SPACECRAFT
17,608 #	17,680 #
113 #	113 #
17,561 #	17,567 #
4,419 #	4,422 #
1,525 #	1,525 #
11,667 #	11,670 #
2,824 #	2,827 #
70 #	70 #
238 ft²	238 ft²
NONE	NONE
GAS THRUSTER	GAS THRUSTER
MINIMUM	NOMINAL
—	—
—	—
—	—

 CONFIG. "E"	 CONFIG. "F
INTERIOR IDENTICAL TO "B." EXTERIOR SIMILAR TO "C" EXCEPT TUBULAR TRUSS STRUCTURE IS REPLACED BY AN INVERTED CONICAL ADAPTER STRUCTURE. SOLAR ARRAY PLATFORM IS INTERRUPTED AT THE HIGH GAIN ANTENNA STOWAGE AREA.	SIMILAR TO "E" EXCEPT THAT S IS SUPPORTED BY UPRIGHT 30° ADAPTER. RETRO-ENGINE IS SU AFT SKIRT BY AN UPRIGHT CO AT HEAD END BY AN INVERTE
17,686 #	17,726 #
376 #	376 #
17,350 #	17,390 #
4,279 #	4,296 #
1,525 #	1,525 #
11,596 #	11,619 #
2,684 #	2,701 #
70 #	70 #
200 ft²	238 ft²
NONE	NONE
MILD DETONATING FUSE	MILD DETONATING
MINIMUM	MINIMUM
—	—
—	—
—	—

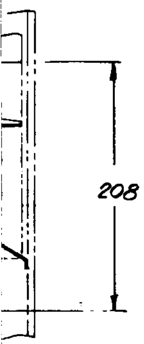
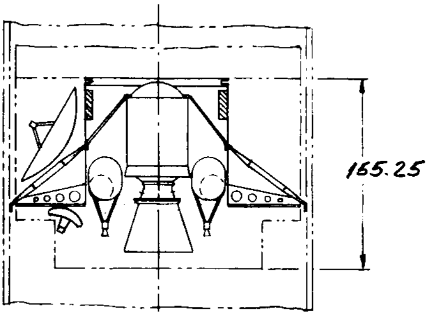
 "	 CONFIG. "G"
ACECRAFT CONICAL PPORTED AT NE AND D CONE.	SIMILAR TO "D" EXCEPT RETRO-ENGINE IS SUPPORTED BY AN UPRIGHT CONE STRUCTURE.
	17,289#
	113#
	17,176#
	4,016#
	1,617#
	11,863#
	2,312#
	70#
	238 ft^2
	NONE
FUSE	GAS THRUSTER
	NOMINAL
	22.5 C.P.S.
	13.0 C.P.S.
	15.2 C.P.S.

Figure 3.1-2. Solid Retropropulsion Configurations

The Midcourse and Orbit Adjust System basically consists of four monopropellant cylindrical tanks which each have their own thrust chamber and two pressurization spherical tanks. These components are attached to the basic cylindrical structure by individual trusses, forming a separate modular structure devoted entirely to this system. The solar array is comprised of 15 trapezoidal fixed fiberglass substrate panels to which the solar cells are attached. These panels attach to 16 radial ribs which in turn are assembled to a short section of the basic 120-inch diameter cylindrical shell structure. This structure houses the Attitude Control System (which is two redundant assemblies or gas tanks), regulators, and plumbing to the nozzles. Each system may be removed from the structure for sterilization.

Major components also mounted on this module, consist of the planet scan platform and 7 1/2-foot diameter high-gain antenna, medium-gain antenna, relay antenna, low-gain antenna, Sun sensors, magnetometer on a boom (in flight electronic disconnect), and launch antenna. Environmental shields are provided at each end of the cylindrical Spacecraft body structure, providing meteoroid protection and also thermal insulation. An emergency separation system of gas-operated separation nuts is provided at the Flight Capsule interface.

3.2 Transtage General Description. The feasibility of utilizing the Titan III-C Transtage Propulsion System for the VOYAGER Mission was studied and configurations are presented herein for both the modified and unmodified arrangements.

Transtage (Figure 3.2-1) consists primarily of two storable liquid propellant titanium tanks of different volumes arranged side-by-side feeding two gimballed thrust chambers. A cylindrical structural aluminum skirt 10 feet in diameter and 18 inches long serves as the support structure for the tanks and the engine thrust mounts. Initially, the Transtage control module was considered for the housing of the VOYAGER electronic equipment; however, this concept did not compare favorably with the modular toroidal structure recommended for the other configurations and the idea was dropped.

3.2.1 Transtage Characteristics Which Act as Restraints on Configuration Design

a. The propellent acquisition method, which is an attractive passive system, gives the Transtage tankage its characteristic shape. All modifications to the system considered, retain this favorable feature.

b. More than adequate volume is provided in the existing fuel tanks for both orbit insertion and midcourse and orbit adjust requirements. Additional tankage, therefore, is not required for the latter system.

c. The existing fuel tanks have been designed with a margin of safety of 1.8. Skin thickness will have to be increased to comply with the JPL requirement of a margin of safety of 2.2.

d. Roll control is provided by the two gimbaled chambers. The thrust level of these chambers is excessive for the Midcourse and Orbit Adjust System, therefore, separate chambers must be provided.

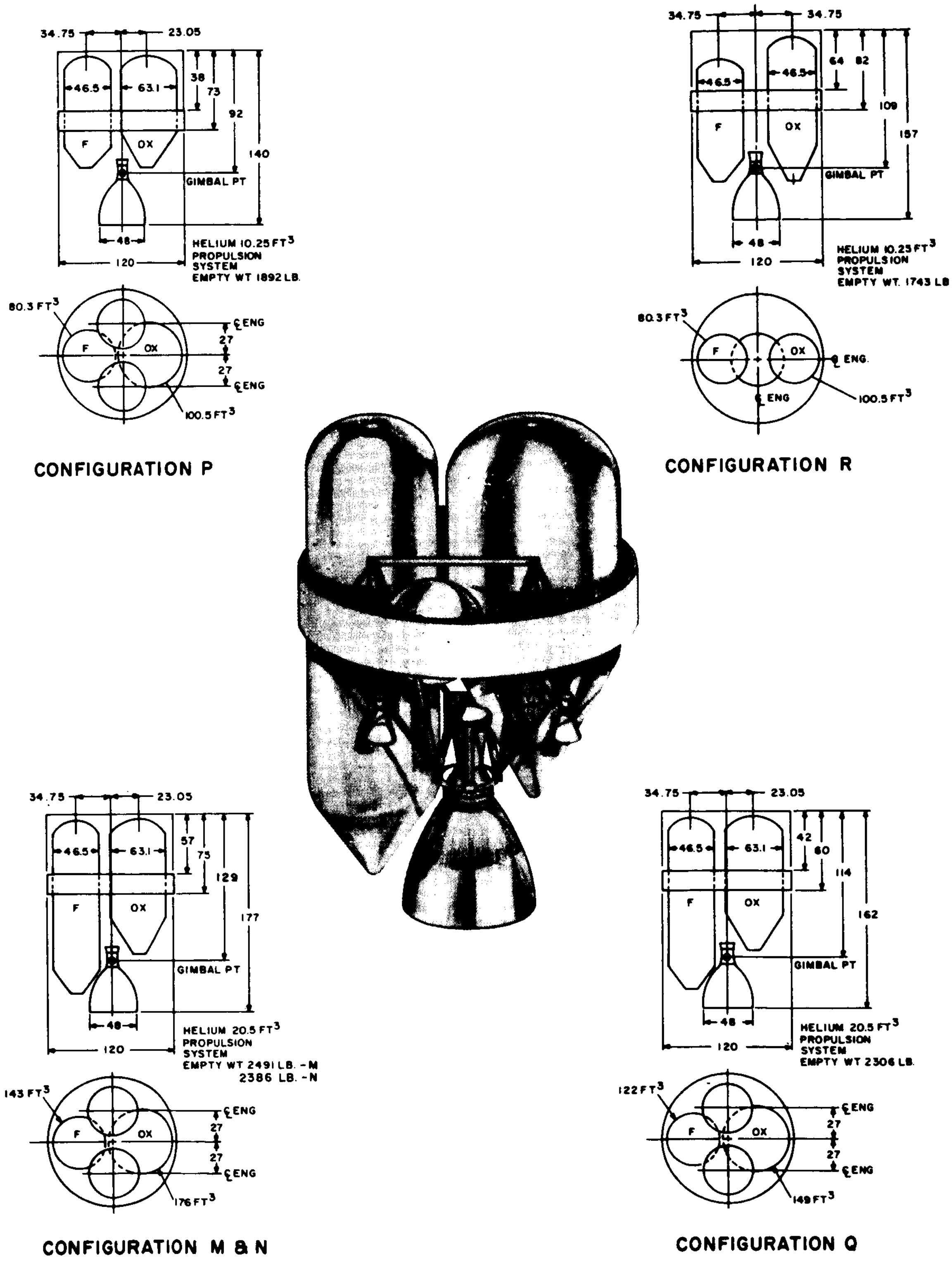


Figure 3.2-1. Transtage

e. The side by side tank arrangement of the existing system forces the electronic packaging torus to a larger dimension than the preferred 120-inch inside diameter.

f. The existing structure provides ideal interface rings at the 120-inch diameter for attachment to other modular system units.

g. There is a lateral cg shift as propellent is used caused by the mixture/density ratio.

3.2.2 Resulting Configurations. The major alternatives influencing the configuration were:

a. With the unmodified versions, the design and shape of the Planetary Vehicle adapter (inverted cone versus upright cone, integrated truss versus combination truss/cone, cone angle).

b. Size, volume, and location of the fuel tanks on the modified versions.

The more promising configurations developed are presented with brief descriptions in Figure 3.2-2.

3.2.3 Configuration Selection. The primary reasons for selection of configuration N over M are:

a. It offers a more rigid structural installation in the shroud. The 45-degree semi-monocoque adapter ties into the vehicle near the Spacecraft cg.

b. Vehicle separation occurs at 162-inch diameter allowing adequate space for the lower portion of the propulsion system to emerge without interference. In addition, the vehicle separation interface affords hard points for the science package and antenna mountings during boost environment.

Several modifications of the Transtage Propulsion System were considered with the primary objective of minimizing the overall length under the shroud. The versions considered ranged from shortening the tanks by simply removing the upper tank barrels representing a minimum modification, to a four equal-sized tank arrangement using only the fuel tank upper and lower domes to contain both fuel and oxidizer and using only a single engine. This latter modification, however, was considered to involve major redesign and, consequently, was discarded even though it utilized existing Transtage hardware. Those configurations which were retained, P, Q, and R are illustrated in Figure 3.2-2. Each of the configurations utilizes a blowdown type of pressurization system supplying midcourse, orbit insertion and orbit adjust performance requirements, and are propellant sized for the 1975/77 Mission requirements.

Configuration Q represents the simplest modification which consists of shortening both the fuel and oxidizer tanks by 18 inches. Configurations P and R have no significant advantages relative to Q, and hence Q was selected as the modified Transtage configuration to be compared with the other systems.

3.2.4 Description of Selected Configurations. The preferred Transtage configurations (unmodified and modified), Figure 3.2-3, are essentially the same with the exception of the reduced equipment module and tankage length on the modified version. The basic body structure consists of a cylindrical shell with 16 longerons and rings at three manufacturing joints. The electrical equipment is housed in 16 integrated assemblies in a toroidal upper structure adjacent to the flight capsule interface. The assembled structure may be considered as a module complete with harnessing, and assembles to the propulsion module via a 120-inch diameter manufacturing joint. The propulsion unit structure is a straight forward cylindrical shell with rings and longerons picking up the longerons of the adjacent structure. The solar array modular structure contains the attitude control system which may be removed as a unit for sterilization. Fifteen solar array fiberglass substrate panels are employed for mounting the solar cells and are supported on 16 radial ribs attached to a 120-inch diameter cylindrical section.

The Spacecraft loads are transmitted to the booster via a conical honeycomb adapter supporting the Spacecraft near the solar array surface at 162-inch diameter, and short truss tubes to the upper ring of the propulsion unit. A micrometeoroid/thermal shield is provided to cover the extremities of the tanks and engine gimbal mechanism. Four midcourse and orbit adjust nozzles are provided, supported from the lower propulsion module ring. Pressurized fuel feed is provided by a blowdown system with a 180-pound weight saving over the existing system with increased reliability. This configuration offers growth potential because of the large ullage volume available.

The antenna and planet scan platform are supported during boost adequately by truss structures which rigidly tie-in at the adapter interface and body structure; no problems are foreseen in this area.

3.3 Lunar Excursion Module Descent Engine. This LEMDE propulsion system consists of two pairs of approximately spherical tanks arranged symmetrically for fuel and oxidizer, feeding a gimbaled throttlable chamber which has a thrust range of 1050 to 10,500 pounds. Fuel expulsion is by high pressure helium in a spherical tank feeding each pair of tanks. The system is mounted in a box beam cruciform structure of aluminum alloy construction, which also serves as landing gear support structure, launch platform, and primary structural support of the Lunar Excursion Module of the Apollo Mission. Geometry, tank sizes and volumes are presented in Figure 3.3-1. A repackaged LEMDE, Figure 3.3-2, was also considered.

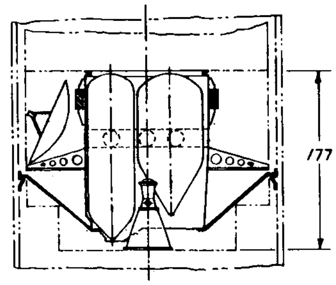
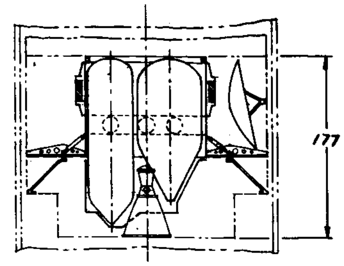
3.3.1 LEMDE Characteristics Which Act As Restraints On Configuration Design

a. The gimbaled engine with its throttling capability, and adequate tankage volume, provide a combined system for the midcourse, orbit injection, and orbit adjust requirements of the VOYAGER Mission. Therefore, no additional tankage is necessary.

b. There is no capability provided for propellent acquisition; therefore, it is proposed to accomplish this with screens in the tanks.

c. The single engine cannot provide roll control capability.

d. There is a very small distance between the cg of the system and the gimbal point of the chamber. As the propulsion system weight is of the order of three times the weight of combined Flight Spacecraft and Flight Capsule a severe autopilot control

Propulsion Selection Matrix Transtage Configurations	 Unmodified "M"	 Unmodified "N"
Design Features	The basic body struct. consists of a cylindrical shell with 16 longerons & rings at the 3 mfg. joints. The electronic equipment is housed in the upper toroidal ring struct. in 16 bays. The latter modular structure assembles to the transtage propulsion system structure, modified to incorporate the 16 longerons. The solar array module structure containing the attitude control system comprises 15 trapezoidal substrate panels supported by 16 radial ribs attached to a short cylindrical section. Between this section and the adapter is a further cylindrical section, which doubles as a thermal shield and a micrometeoroid barrier. An inverted cone honeycomb adapter is employed.	Similar to "M" except that a conical honeycomb adapter supports the vehicle at 162" diameter near the solar array surface, and loads are carried through to the lower forged ring of the propulsion unit by the short truss tubes.
Total Earth Take-off Weight	19,665#	19,350#
Adapter - S/C to Booster - Weight	281#	357#
Total Cruise Weight	19,384#	18,993#
Burn-out Weight	5,212#	4,880#
Propulsion Sys. Total Wt. - Empty	2,549#	2,386#
Propulsion Sys. Total Wt. - Loaded	13,721#	13,605#
Bus Weight	2,557#	2,388#
Environmental Shield Weight	106#	106#
Solar Array Area - Fixed	236 ft²	236 ft²
Solar Array Area - Deployed	None	None
Recommended Separation System	Mild Detonating Fuse	Gas Thruster
Plume Impingement	Minimum	Minimum
Natural Frequency - First Mode: Longitudinal	13.2 C.P.S.	(16.0 C.P.S.)
Natural Frequency - First Mode: Lateral	8.1 C.P.S.	(13.0 C.P.S.)
Natural Frequency - First Mode: Torsional	17.0 C.P.S.	(18.0 C.P.S.)

Notes:-

1. Propellant weights are based on:
   Bus weight = 2500#
   Capsule weight = 3000#
2. Natural frequencies shown in parenthesis are estimates.

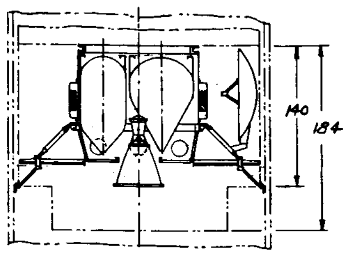
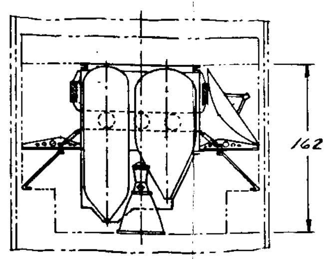
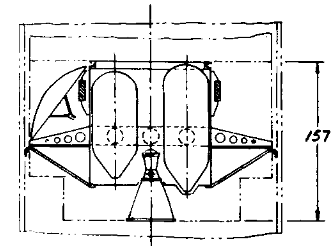
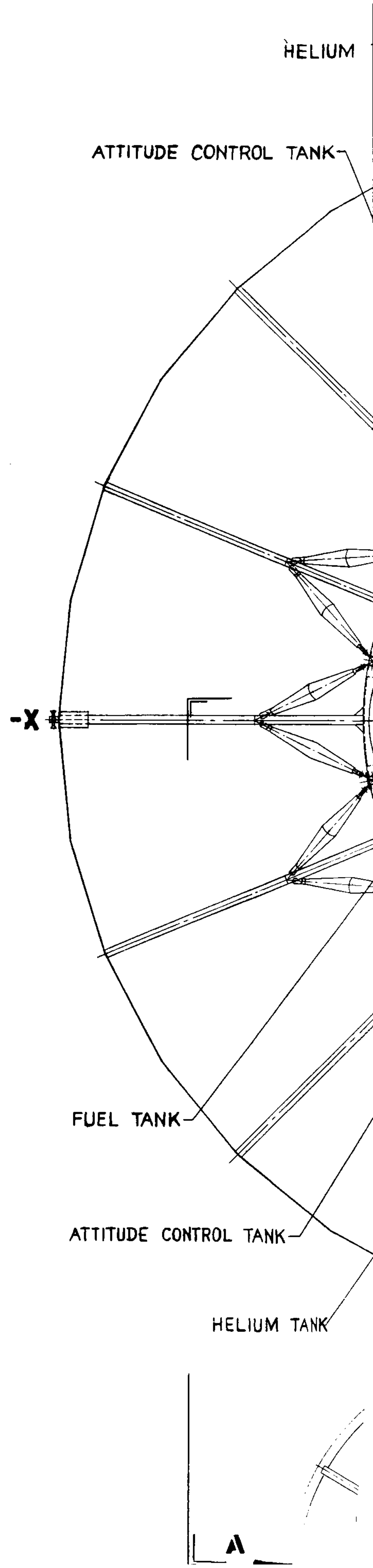
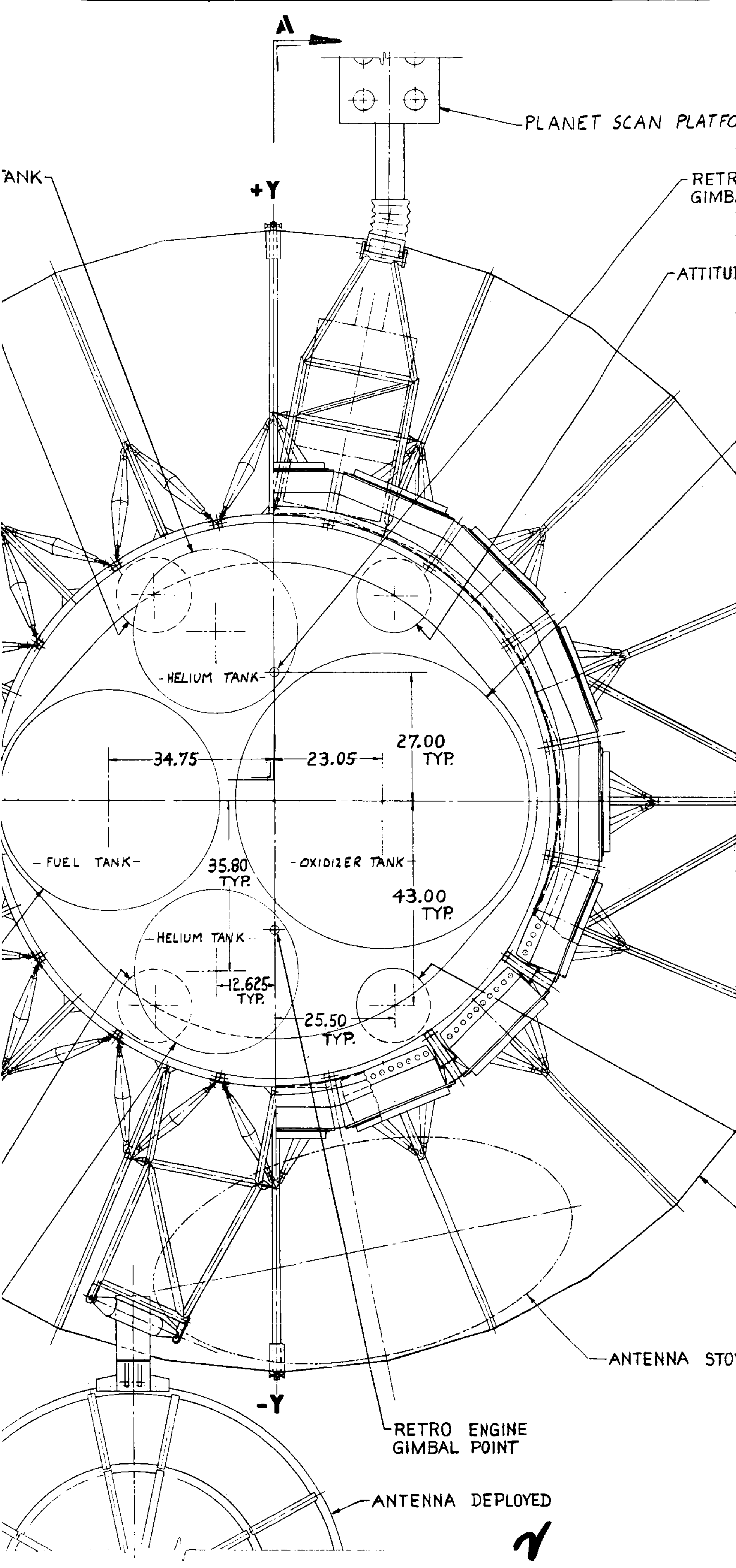
		
MODIFIED "P"	MODIFIED "Q"	MODIFIED "R"
THE CYLINDRICAL SECTION OF THE PROPULSION UNIT STRUCTURE HAS BEEN ELIMINATED DUE TO THE EXTREME REDUCTION OF TANK LENGTH, WHICH IS STILL OF ADEQUATE VOLUME FOR THE '75/'77 MISSION. A SHORT HONEYCOMB ADAPTER IS EMPLOYED, SUPPORTING THE VEHICLE AT 200" DIAMETER. LOADS ARE CARRIED THRU TO THE LOWER RING OF THE EQUIPMENT MODULE VIA TRUSS TUBES. THE REDUCED TANK LENGTH RESULTS IN LESS MICROMETEORITE SHIELDING & MORE SOLAR ARRAY AREA.	IDENTICAL TO "N" WITH THE EXCEPTION OF THE SHORTENED EQUIPMENT MODULE LENGTH ALLOWED BY THE REDUCTION OF CYLINDRICAL TANK LENGTH.	BY UTILIZING A FUEL TANK FOR OXIDIZER, TANKS OF EQUAL DIAMETER ARE EMPLOYED IN THIS CONFIGURATION PROVIDING SPACE FOR A SINGLE ENGINE IF REQUIRED. STRUCTURE IS SIMILAR TO "M" EXCEPT A NEW PROPULSION UNIT STRUCTURE MUST BE PROVIDED. AN INVERTED 60° CONICAL ADAPTER OF HONEYCOMB CONSTRUCTION IS EMPLOYED, PROVIDING THE SHORTEST CONFIGURATION.
18,503#	18,233#	17,753#
140#	357#	300#
18,363#	17,876#	17,453#
4,620#	4,753#	4,266#
2,072#	2,366#	1,743#
12,815#	12,396#	11,930#
2,508#	2,387#	2,478#
40#	93#	45#
250 ft^2	236 ft^2	236 ft^2
NONE	NONE	NONE
GAS THRUSTER	GAS THRUSTER	MILD DETONATING FUSE
HIGH	MINIMUM	MINIMUM
(16.0 C.P.S.)	(17.0 C.P.S.)	12.2 C.P.S.
(14.0 C.P.S.)	(15.0 C.P.S.)	10.0 C.P.S.
(18.0 C.P.S.)	(19.0 C.P.S.)	21.3 C.P.S.

Figure 3.2-2. Transtage Configurations Developed

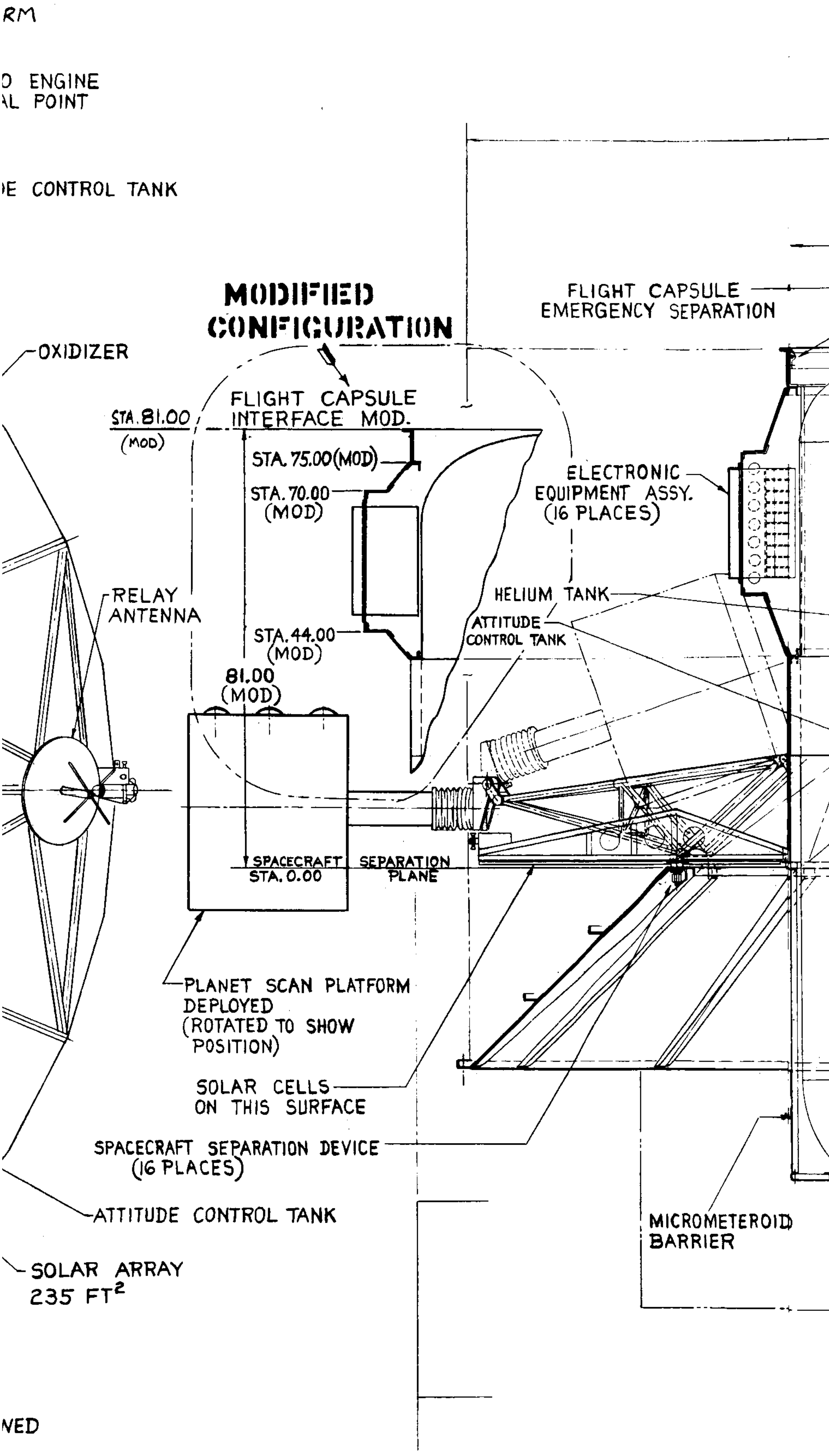


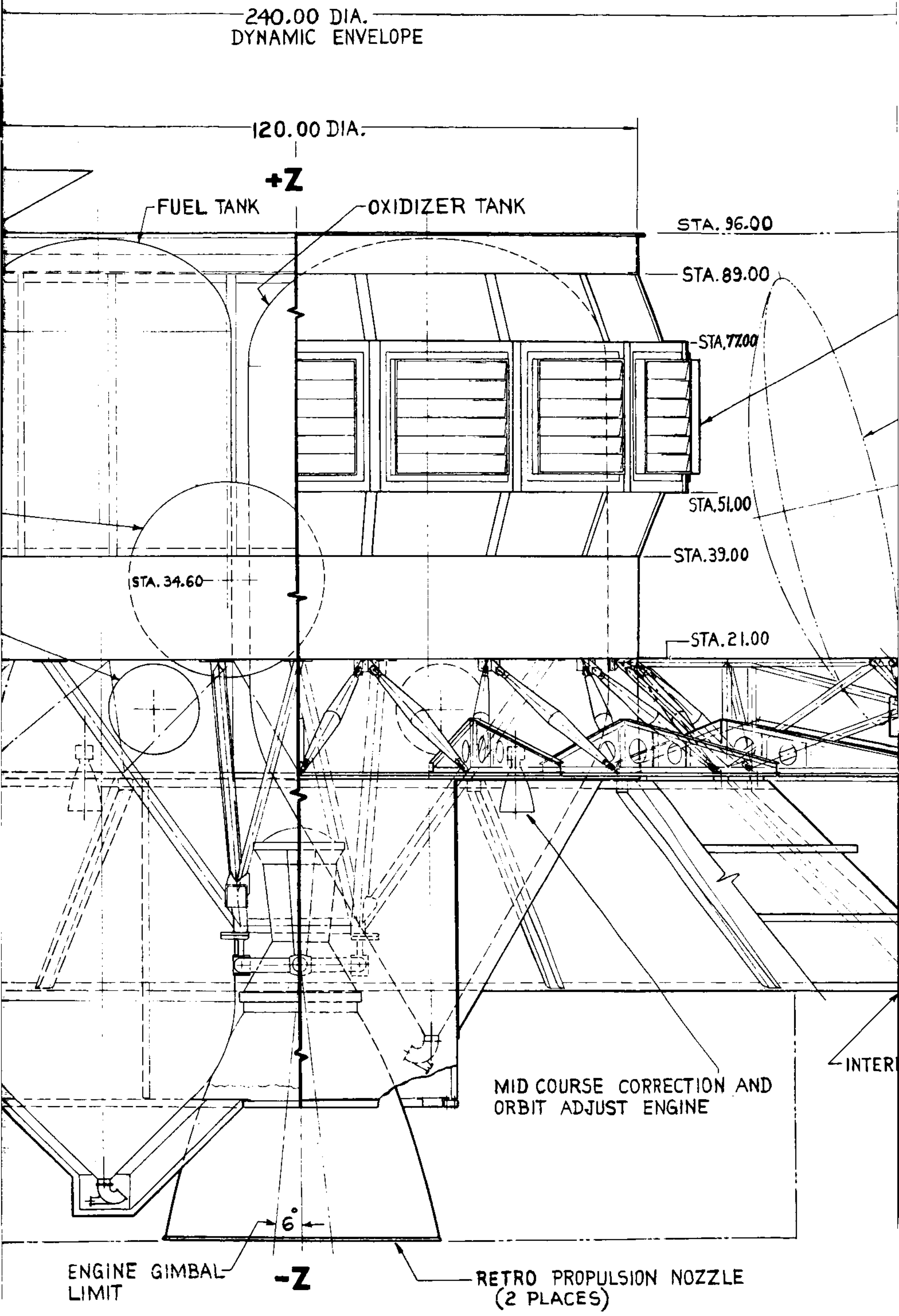
HELIUM
ATTITUDE CONTROL TANK
-X
FUEL TANK
ATTITUDE CONTROL TANK
HELIUM TANK
A

1



A
PLANET SCAN PLATFO
ANK
RETR
GIMB
ATTITU
+Y
-HELIUM TANK-
34.75
23.05
27.00
TYP.
- FUEL TANK-
35.80
TYP.
- OXIDIZER TANK-
43.00
TYP.
-HELIUM TANK-
12.625
TYP.
25.50
TYP.
ANTENNA STO
-Y
RETRO ENGINE
GIMBAL POINT
ANTENNA DEPLOYED


ENGINE
POINT
CONTROL TANK
MODIFIED CONFIGURATION
FLIGHT CAPSULE EMERGENCY SEPARATION
OXIDIZER
FLIGHT CAPSULE INTERFACE MOD.
STA. 81.00 (MOD)
STA. 75.00 (MOD)
STA. 70.00 (MOD)
ELECTRONIC EQUIPMENT ASSY. (16 PLACES)
RELAY ANTENNA
HELIUM TANK
ATTITUDE CONTROL TANK
STA. 44.00 (MOD)
81.00 (MOD)
SPACECRAFT STA. 0.00
SEPARATION PLANE
PLANET SCAN PLATFORM DEPLOYED (ROTATED TO SHOW POSITION)
SOLAR CELLS ON THIS SURFACE
SPACECRAFT SEPARATION DEVICE (16 PLACES)
ATTITUDE CONTROL TANK
MICROMETEOROID BARRIER
SOLAR ARRAY 235 FT²



SECTION A-A
(ROTATED 90° C.C.W)

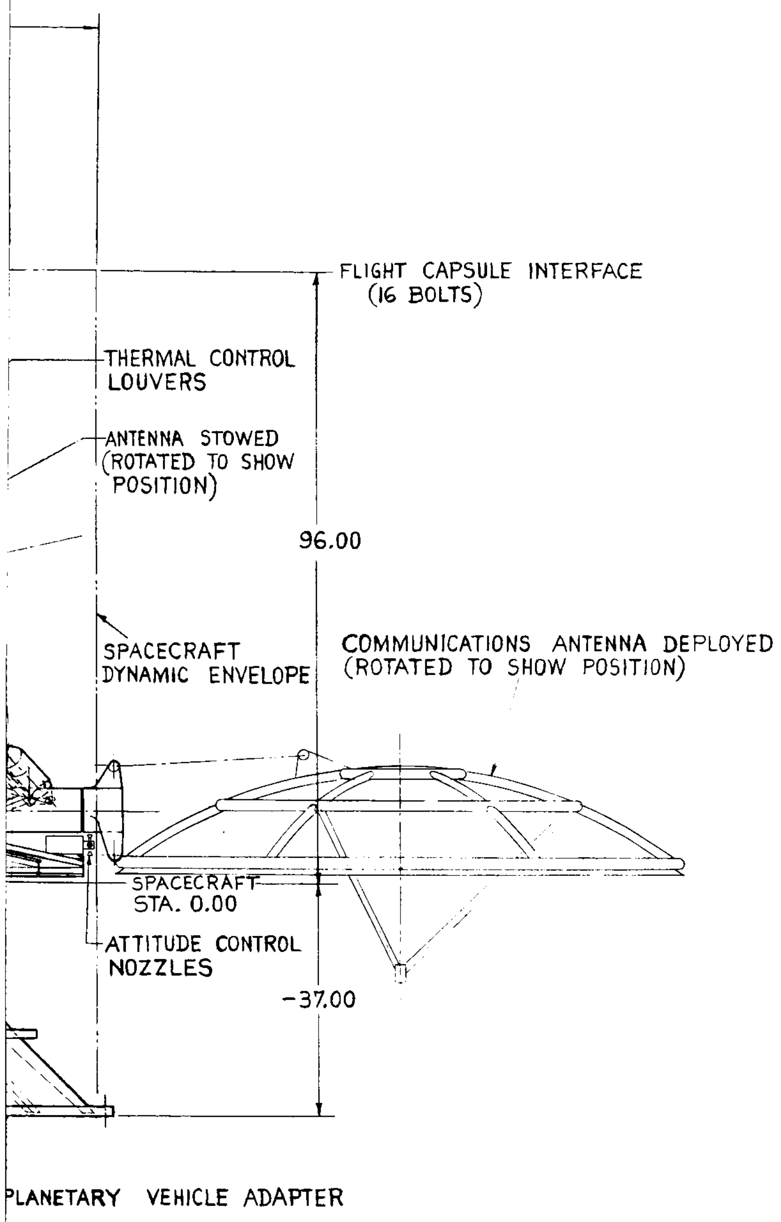


Figure 3.2-3. Preferred Transtage Configurations

6

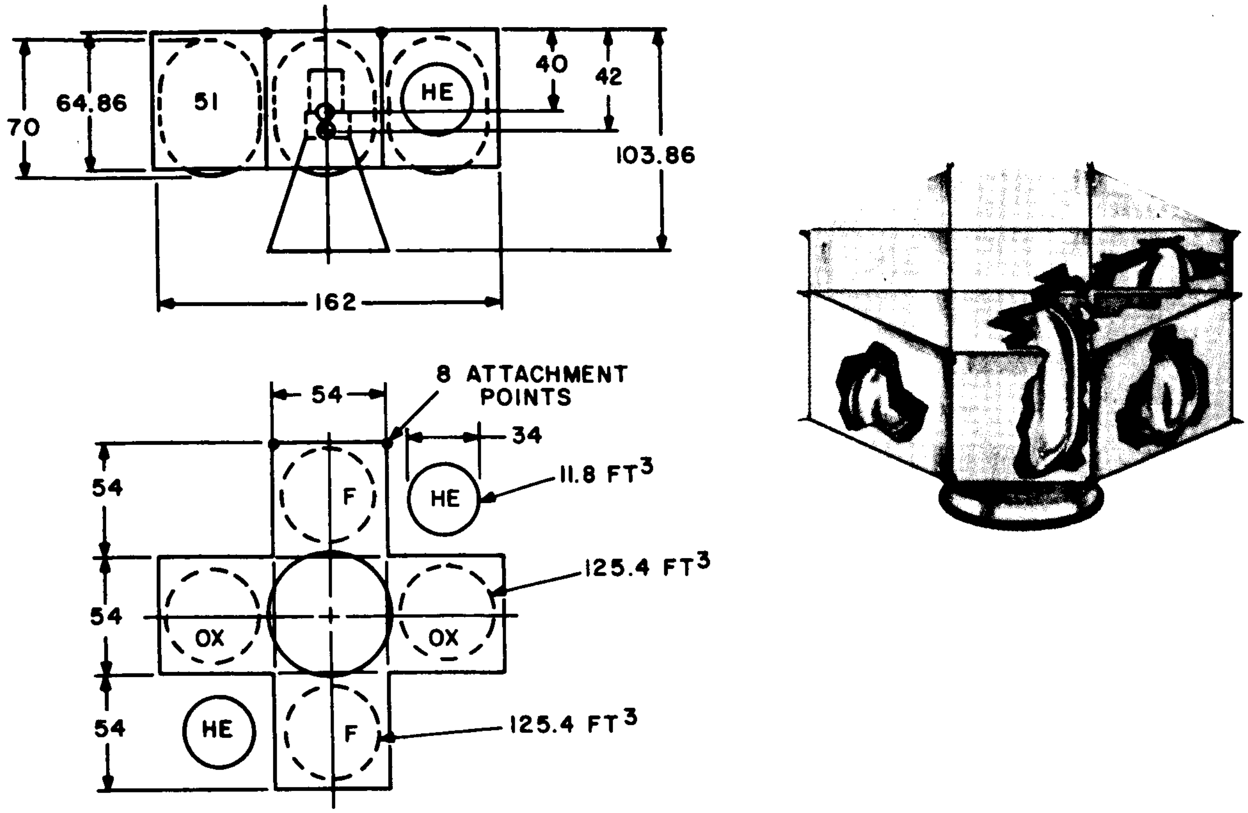


Figure 3.3-1. Unmodified LEMDE, Propulsion Schematic

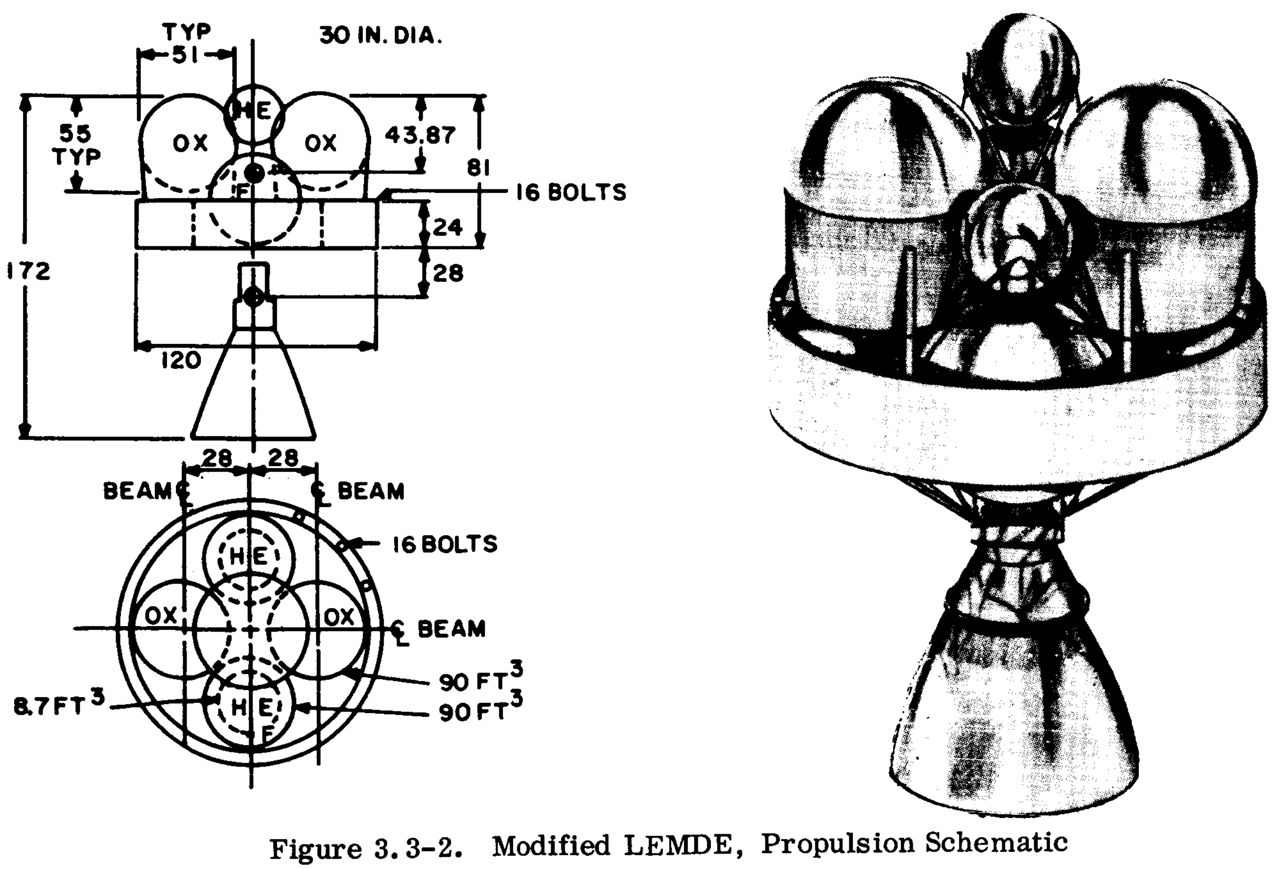


Figure 3.3-2. Modified LEMDE, Propulsion Schematic

problem arises with the resulting short control arm between the total Spacecraft cg and the gimbal point.

e. There is insignificant cg travel in X and Y directions as propellent is used, due to the symmetrical tank arrangement.

3.3.2 Resulting Configurations. Three configurations were studied using the LEMDE unmodified. Also an attractive version is presented which is modified to the extent of repackaging the existing system in a new structure. These configurations with brief descriptions are presented in Figure 3.3-3.

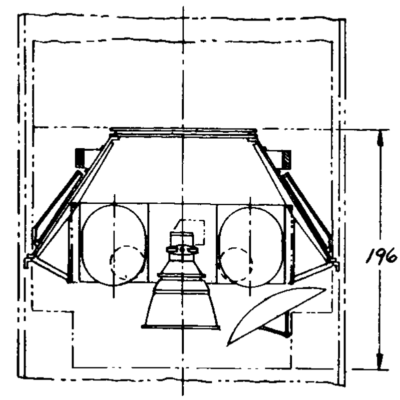
3.3.3 Configuration Selection. The large projected area of the LEMDE structure looking along the thrust axis presents serious blockage to the rejected heat path for the preferred fixed solar array and considerable loss of solar cell efficiency (of the order of 40%) results. Therefore, an array using deployable panels has been designed for configurations H, J, and K.

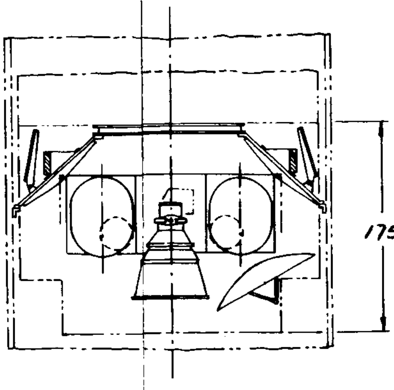
It is now possible on these configurations to locate antennas, etc., on the sun side of the Spacecraft, since no array shading problem exists. Also an optimum lightweight structure may be designed to provide the dual functions of Spacecraft structure and Planetary Vehicle adapter (Configurations H and J). Configuration K attempts to utilize the propulsion system structure as spacecraft structure, but requires a short inverted cone adapter. Configuration J is basically the same as H except that the Spacecraft structural cone has been shortened, saving structural weight and shroud length. However, the solar array panels when folded in the shroud, cover the electronic equipment module. This is considered a significant problem from the point of view of ground cooling, therefore, configurations J and K were eliminated. Configuration H also has increased structural stiffness, and would be the recommended design if no modification to the structure could be tolerated.

Significant weight saving and considerably improved overall Spacecraft design is possible by repackaging the system components in a new structure. This led to the modified configuration L which essentially solves the aforementioned autopilot and solar array problems. In addition, it allows a significant weight reduction. This modified version was sufficiently attractive to be included in the overall evaluation and selection described in Section 2.0.

3.3.4 Description of Selected Configurations

3.3.4.1 LEMDE Unmodified Configuration H. This configuration, Figure 3.3-4, consists of the existing LEMDE descent vehicle cruciform structure and propulsion system. The cruciform structure is attached to the underside of a load carrying 30-degree conical shell. The lower edge of the conical shell is attached to the Saturn V shroud and the upper edge supports the capsule. The electronic modules are mounted on the outside of the cone near the upper edge. The deployable solar array panels are hinged from the outside of the cone near the lower edge. The modular concept is followed to the greatest extent possible. There is a propulsion module, an electronic module, and the adapter for attachment of the capsule to the Saturn V shroud. The solar array panels, antennas, and sensors are separate installation items.

PROPULSION SELECTION MATRIX L.E.M.D.E. CONFIGURATIONS		 UNMODIFIED "H"
DESIGN FEATURES		1. BASIC STRUCTURE: 30° HONEYCOMB CONICAL SHELL. 2. ELECTRONIC EQUIPMENT IS HOUSED IN TWENTY BAYS IN TOROIDAL MODULAR STRUCT. 3. SOLAR ARRAY HAS SEVENTEEN TRAPEZOIDAL DEPLOYABLE PANELS. 4. PROPULSION UNIT ATTACHES TO FITTINGS ON EIGHT LONGERONS. 5. ANTENNA IS STOWED AT MARS ENCOUNTER ANGLE AND CAN BE USED IF DEPLOY. MECH. FAILS. 6. NO SPACECRAFT/LAUNCH VEHICLE ADAPTER.
TOTAL EARTH TAKE-OFF WEIGHT		20,205#
ADAPTER - S/C TO BOOSTER - WEIGHT		326 #
TOTAL CRUISE WEIGHT		19,879 #
BURN-OUT WEIGHT		5,578#
PROPULSION SYS. TOTAL WT. - EMPTY		3,019 #
PROPULSION SYS. TOTAL WT. - LOADED		14,320 #
BUS WEIGHT		2,519 #
ENVIRONMENTAL SHIELD WEIGHT		40#
SOLAR ARRAY AREA - FIXED		NONE
SOLAR ARRAY AREA - DEPLOYED		229 ft^2
RECOMMENDED SEPARATION SYSTEM		MILD DETONATING FUSE
PLUME IMPINGEMENT		NOMINAL
NATURAL FREQUENCY - FIRST MODE	LONGITUDINAL	21.3 C.P.S.
	LATERAL	12.5 C.P.S.
	TORSIONAL	19.4 C.P.S.

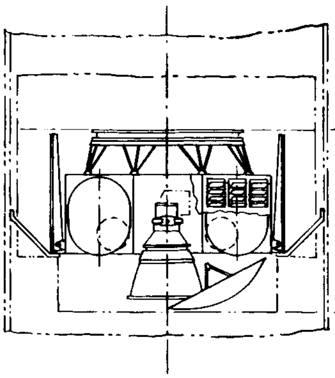


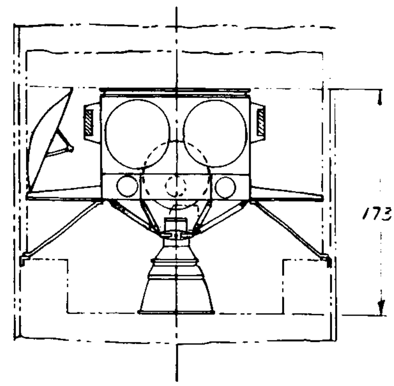
UNMODIFIED "J"

1. BASIC STRUCTURE:- 45° HONEYCOMB CONICAL SHELL.
2. ELECTRONIC EQUIPMENT IS HOUSED IN TWENTY BAYS IN TOROIDAL MODULAR STRUCTURE.
3. SOLAR ARRAY HAS SEVENTEEN TRAPEZOIDAL DEPLOYABLE PANELS.
4. PROPULSION UNIT ATTACHES TO FITTINGS ON EIGHT LONGERONS.
5. ANTENNA IS STOWED AT MARS ENCOUNTER ANGLE AND CAN BE USED IF DEPLOYMENT MECHANISM FAILS.
6. NO SPACECRAFT/LAUNCH VEHICLE ADAPTER.

19,960 #
87 #
19,873 #
5,572 #
3,019 #
14,320 #
2,513 #
40 #
NONE
229 ft²
MILD DETONATING FUSE
NOMINAL
—
—
—

2

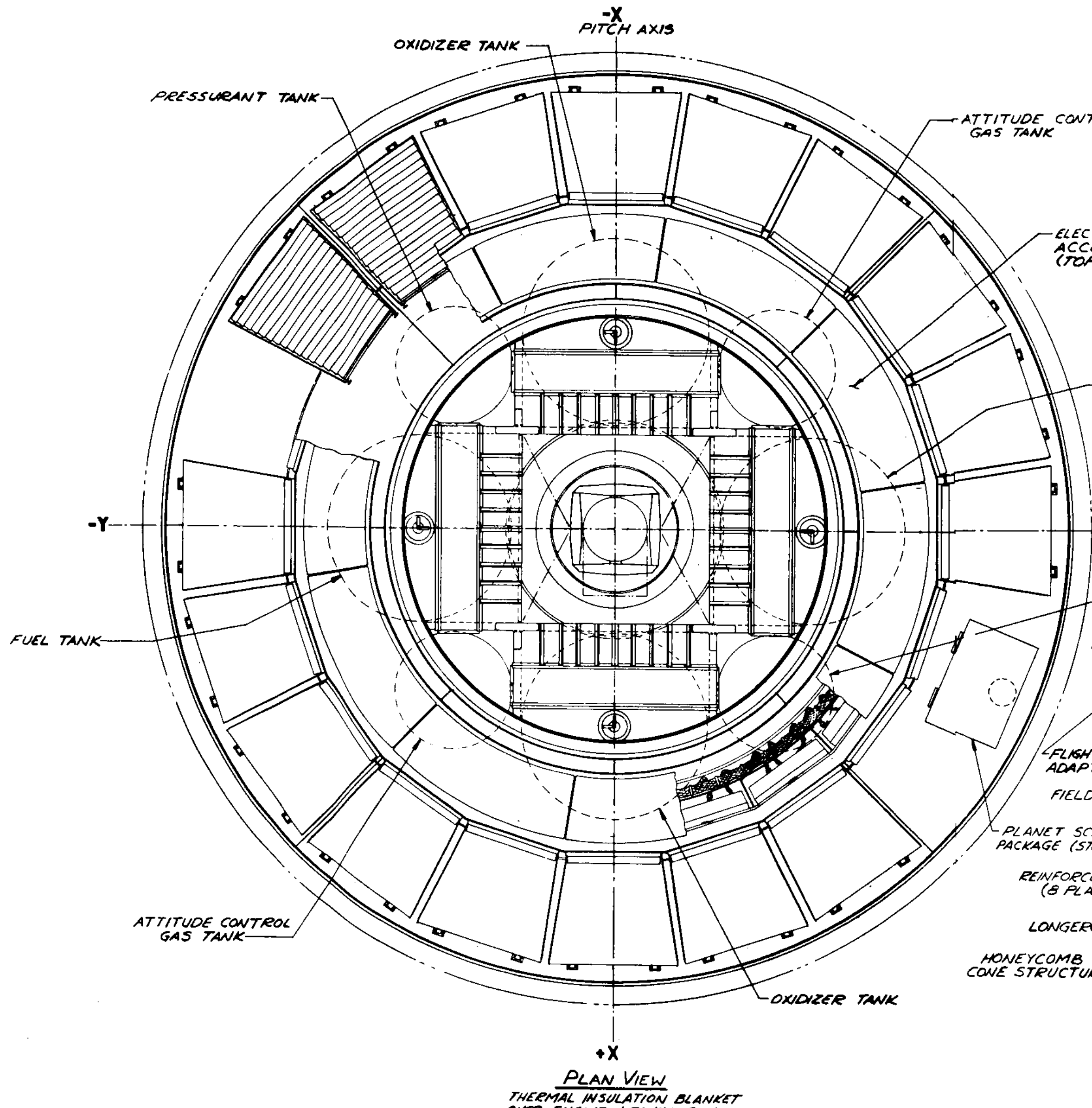




UNMODIFIED "K"	MODIFIED "L"
1. BASIC STRUCTURE:-TRUSS COMBINED WITH L.E.M.D.E. STRUCTURE, SUPPORTED ON INVERTED HONEYCOMB ADAPTER ATTACHED WITH 8 EXPLOSIVE BOLTS 2. ELECTRONIC EQUIPMENT IS HOUSED IN THE DIAGONAL THERMAL METEOROID SHIELD, THREE ASSYS. PER BAY MAKING A TOTAL OF TWELVE. 3. SOLAR ARRAY HAS SEVEN RECTANGULAR DEPLOYABLE PANELS. 4. ANTENNA MUST BE DEPLOYED.	1. BASIC STRUCTURE:- CYLINDRICAL SHEET METAL SHELL WITH LONGERONS AND RINGS AT MANUFACTURED JOINTS. 2. PROPULSION MODULE STRUCTURE IS CRUCIFORM BEAMS WITHIN CYLINDRICAL SHELL. 3. SOLAR ARRAY IS FIXED WITH FIFTEEN FIXED TRAPEZOIDAL PANELS. 4. ELECTRONIC EQUIPMENT HOUSED IN SIXTEEN BAYS IN TOROIDAL MODULAR STRUCTURE. 5. EXHAUST PLUME INPINGEMENT IS MINIMIZED. 6. ADAPTER IS 45° HONEYCOMB CONICAL SHELL.
19,998#	17,638#
205#	307#
19,793#	17,331#
5,492#	4,400#
3,019#	1,984#
14,320#	11,915#
2,433#	2,376#
40#	40#
NONE	218 ft²
250 ft²	NONE
EXPLOSIVE BOLT	MILD DETONATING FUSE
NOMINAL	MINIMUM
———	12.4 C.P.S.
———	9.8 C.P.S.
———	18.0 C.P.S.

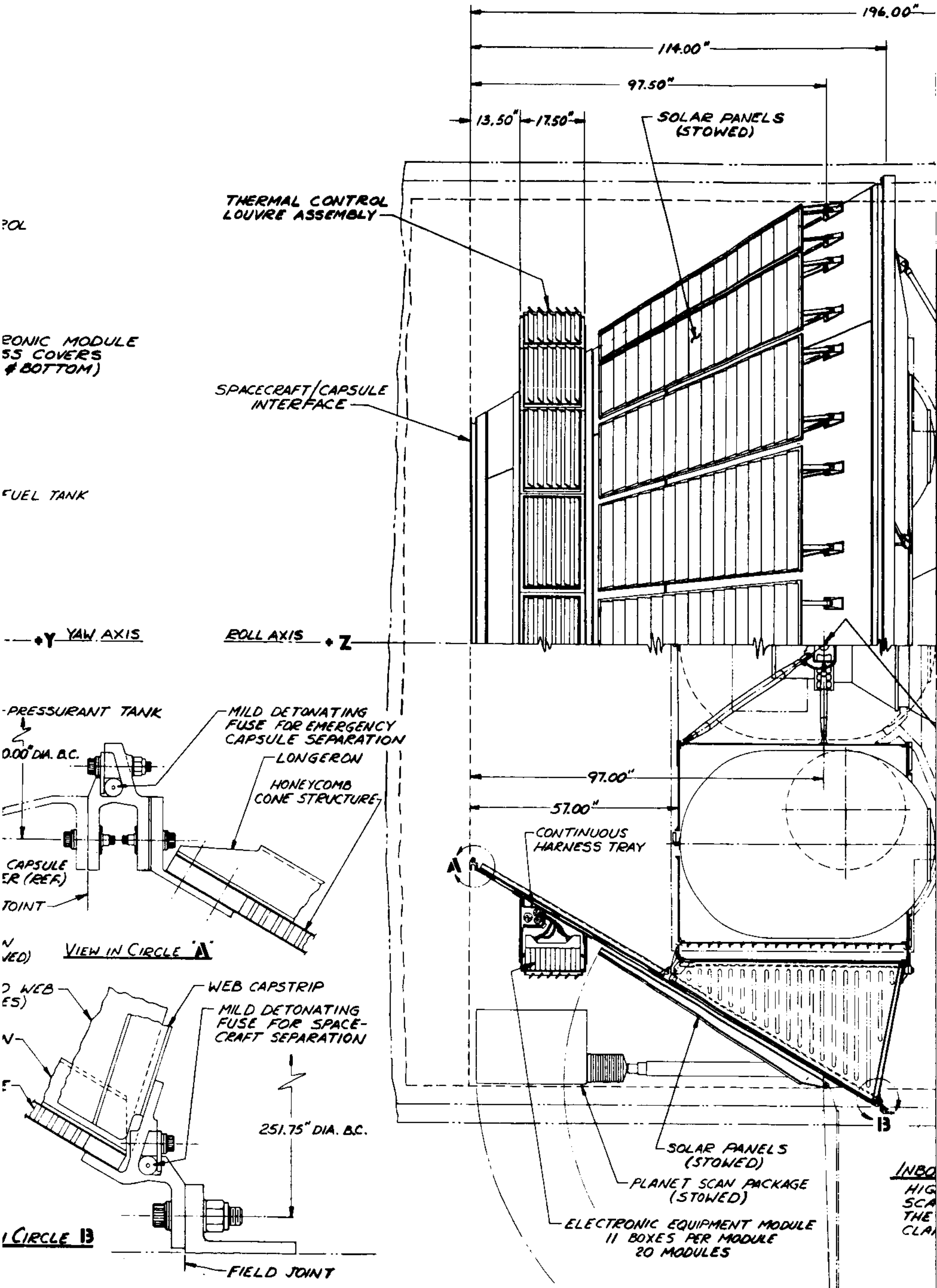
Figure 3.3-3. LEMDE Configurations

3



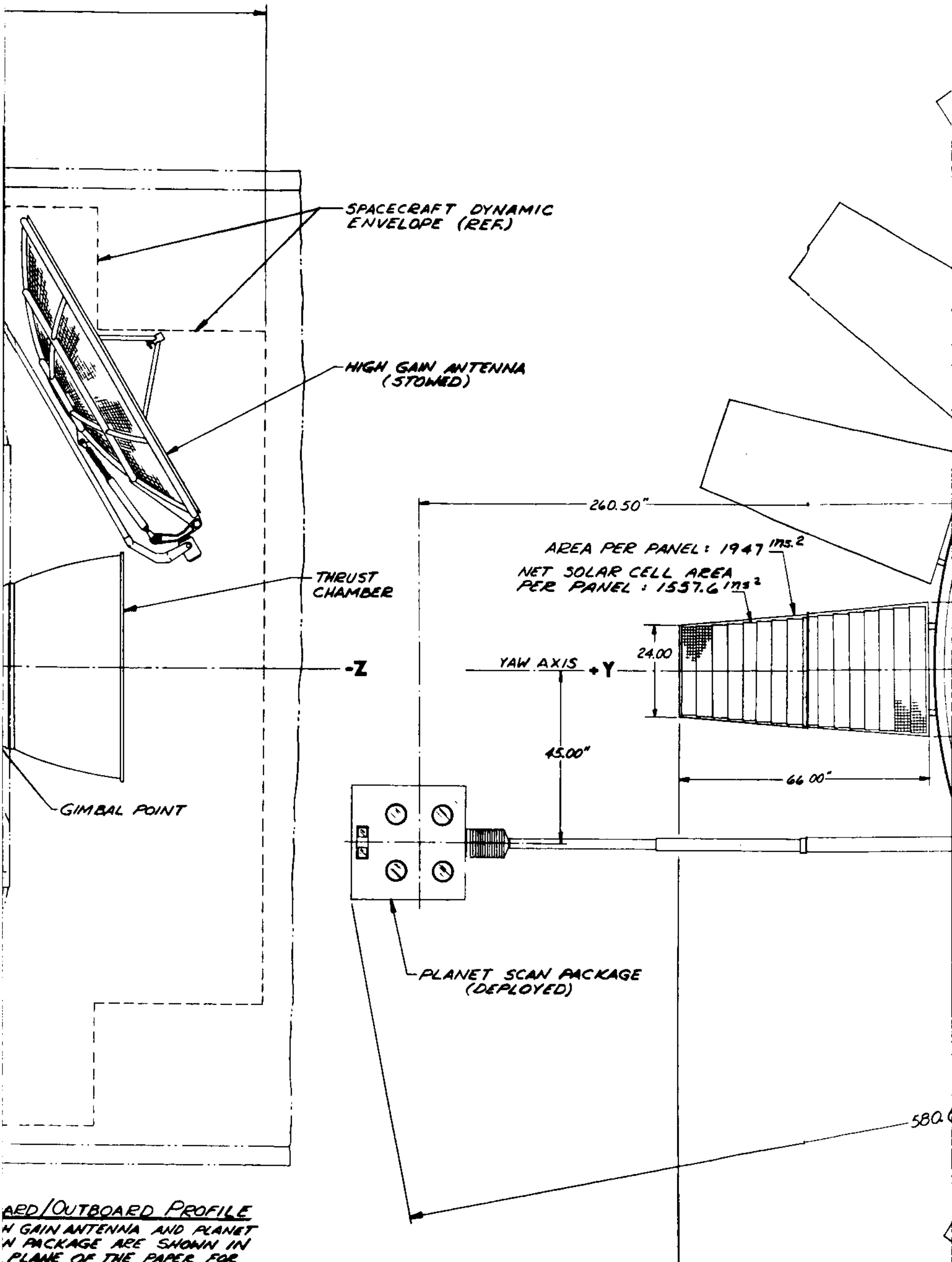
PLAN VIEW
THERMAL INSULATION BLANKET OVER ENGINE & TANKAGE NOT SHOWN

1



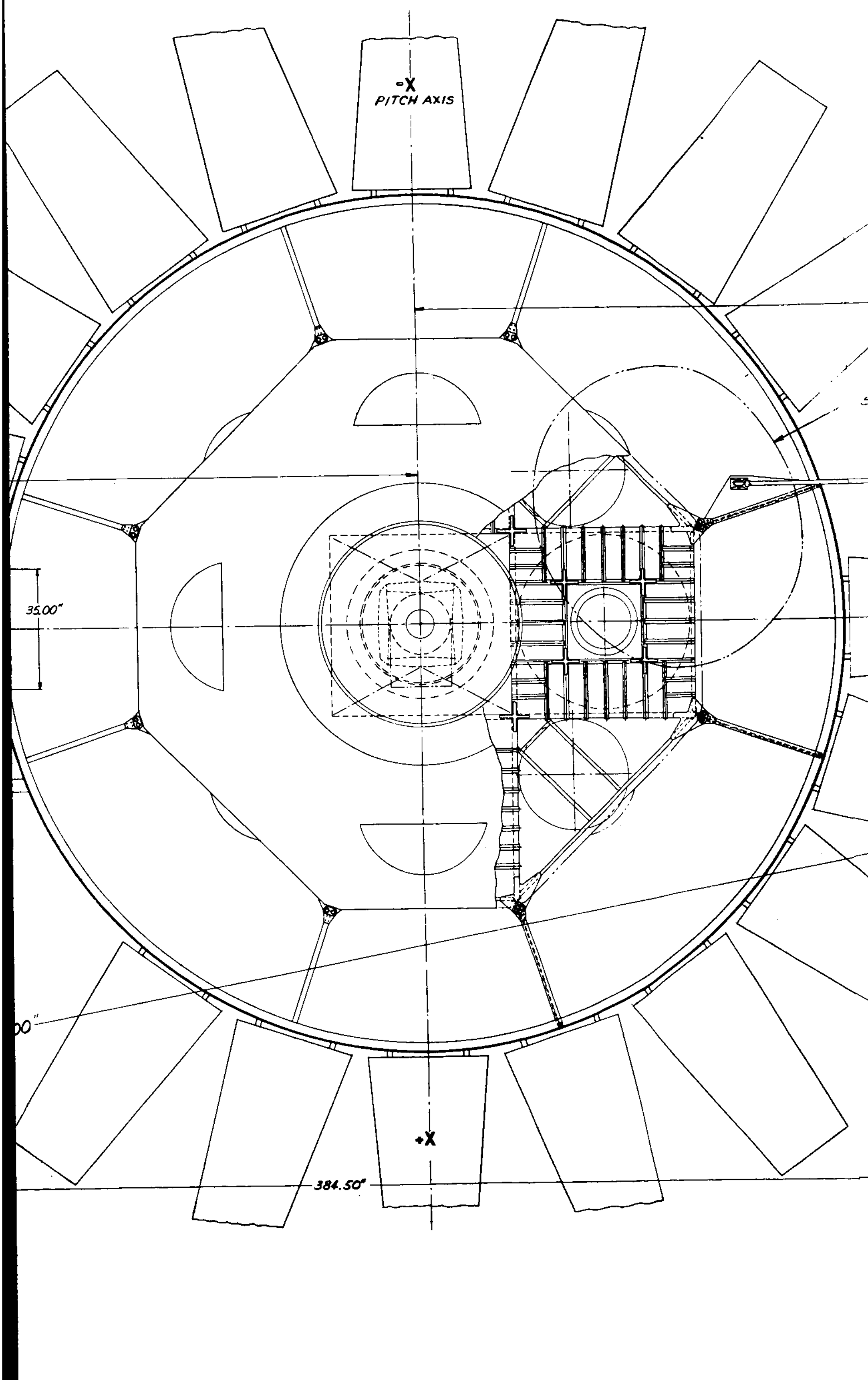
196.00"
114.00"
97.50"
13.50"
17.50"
SOLAR PANELS (STOWED)
THERMAL CONTROL LOUVRE ASSEMBLY
SPACECRAFT/CAPSULE INTERFACE
+Y YAW AXIS
ROLL AXIS +Z
PRESSURANT TANK
MILD DETONATING FUSE FOR EMERGENCY CAPSULE SEPARATION
LONGERON
HONEYCOMB CONE STRUCTURE
VIEW IN CIRCLE 'A'
WEB CAPSTRIP
MILD DETONATING FUSE FOR SPACE-CRAFT SEPARATION
251.75" DIA. B.C.
FIELD JOINT
97.00"
57.00"
CONTINUOUS HARNESS TRAY
A
B
SOLAR PANELS (STOWED)
PLANET SCAN PACKAGE (STOWED)
ELECTRONIC EQUIPMENT MODULE 11 BOXES PER MODULE 20 MODULES

4 2



ARD/OUTBOARD PROFILE
H GAIN ANTENNA AND PLANET
N PACKAGE ARE SHOWN IN
PLANE OF THE PAPER FOR
RIFICATION PURPOSES.

3



~~6~~ 4

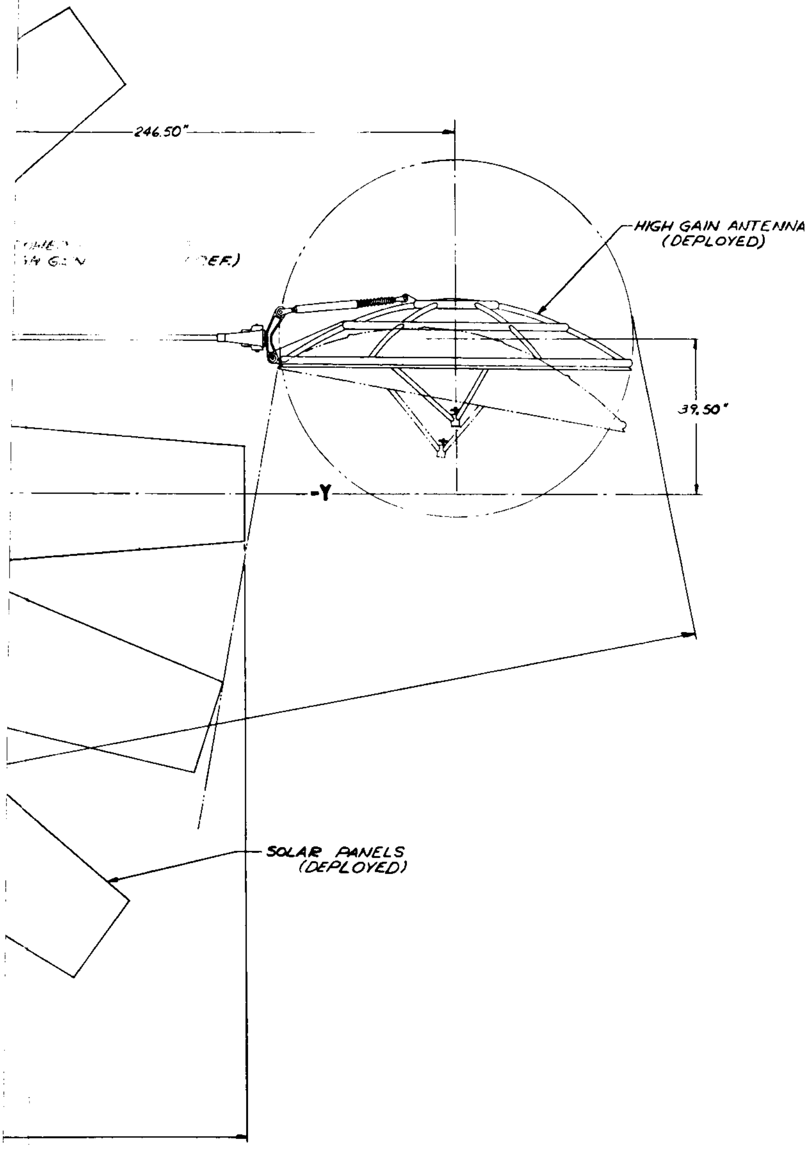


Figure 3.3-4. Unmodified LEMDE Configuration

The existing cruciform structure provides adequate hard points for attachment to the underside of the cone. It is possible to use the eight upper landing gear strut fittings with minor modification for this purpose. This provides an excellent connection of the propulsion module to the adapter cone. The cruciform structure is unmodified except for the deletion of fittings and back-up structure for unused landing gear points, water tanks, and similar items. The cruciform is shrouded with a thermal blanket and meteoroid shield in a manner similar to the existing LEMDE descent vehicle.

The adapter cone is a honeycomb structure with eight longerons located to pick up the eight attach points on the upper corners of the propulsion module. The capsule is attached to a uniformly loaded 120-inch diameter closing frame at the top end of the truncated cone. An emergency separation ring is provided so that the Capsule and the Spacecraft may be separated independently of the Capsule Separation System. A MDF ring provides the separation force. The lower edge of the adapter cone is attached to a uniformly loaded adapter frame on the Saturn shroud. Spacecraft/Booster separation is accomplished by MDF ring charge.

The electronic module is located external to the adapter cone providing excellent access capabilities. Radiative thermal transfer to the spacecraft interior is restricted by the adapter cone structure. However, a good conductive path is provided. Twenty electronic packages are used; and the harness is located in an external tray for easy accessibility.

The LEMDE propulsion system is used intact except that the size of the two helium spheres is changed to 34 inches in diameter, the helium pressurization system is revised and new engine gimbal actuators are required.

The solar array panels must be stowed in a vertical position on a series of paddles hinged to the outside of the adapter. They are deployed to an extended position after separation from the Saturn booster.

Storage and deployment of the antenna, planet scanner and other sensors are readily accomplished and present no problems.

The adapter to the Saturn shroud is an inverted honeycomb 45-degree truncated cone. The primary loads are tension on the cone. The propulsion module is attached to the adapter at eight landing gear strut points on the lower corners of the cruciform structure. The attach bolts have explosive nuts that are used for spacecraft separation. The adapter is permanently attached to the Saturn shroud adapter frame in a uniformly loaded fashion and stays with the booster after separation.

The solar arrays are mounted on 17 rectangular paddles, which fold down for deployment. There is space for three more paddles, but this space is used for deploying the high-gain antenna.

3.3.4.2 LEMDE Modified Configuration L. The LEM modified configuration, Figure 3.3-5, consists of a modified LEMDE propulsion system and a new 120-inch diameter structure specifically designed for the VOYAGER Mission. It has many significant advantages over the unmodified configuration. The structure weight will be lighter. Engine plume impingement on the structure and solar array will be virtually eliminated. The solar array panels can be fixed rather than deployable, thereby, increasing reliability. Spacecraft controllability will be improved by increasing the distance between the engine and the Spacecraft cg. Structural growth to the 1975/77 Mission involves gage changes only.

There are five major subassemblies; the Booster/Spacecraft adapter cone, the solar array, propulsion module, electronic module, emergency capsule separation ring. The Booster/ Spacecraft adapter is a uniformly loaded honeycomb cone. The upper end of the cone contains an MDF separation device. The lower edge of the adapter is bolted, with a uniform load distribution, to the booster shroud. The adapter cone weight stays with the Saturn booster, thereby reducing the weight which goes into Mars orbit.

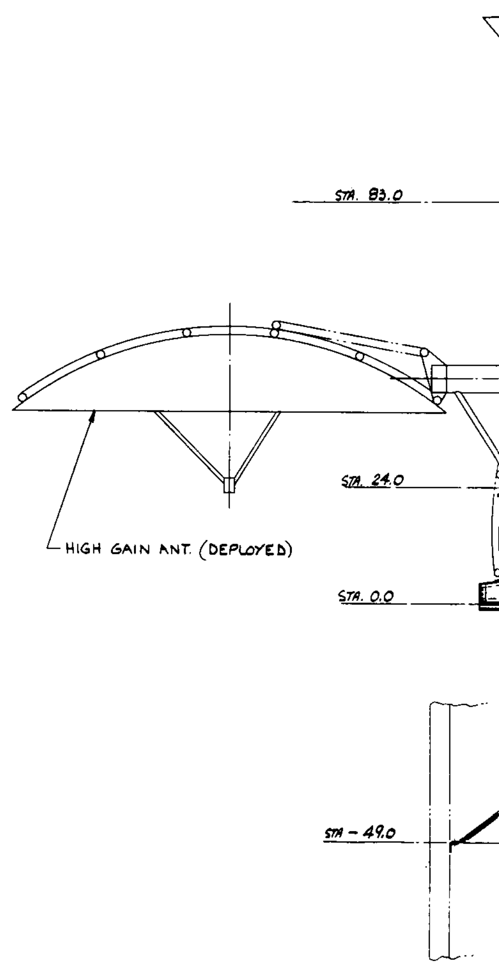
The solar array is located so that sun exposure is very good and radiation of infrared off the back side of the array is unrestricted. The solar array module has 218 square feet of solar cells on 16 panels. These panels are mounted onto 16 ribs to form a separate module which can be attached by 32 bolts to the propulsion module.

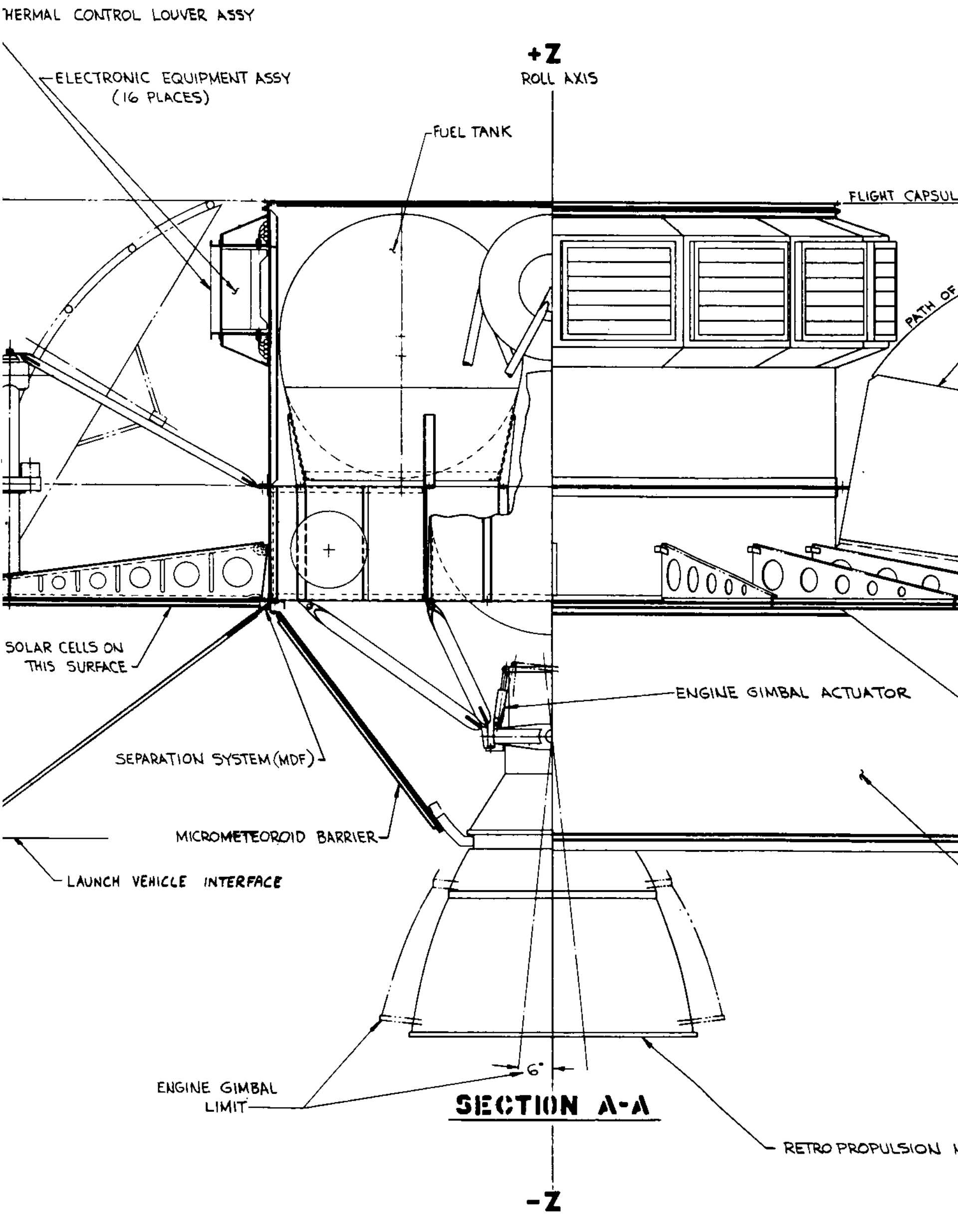
The propulsion module is a 120-inch diameter circular shell 24 inches high containing one centerline cross beam and two auxiliary beams 90 degrees to the centerline beam. The unmodified LEMDE engine is attached by a new engine truss structure to the underside of the beams. The four propellant tanks are made with the 51-inch diameter hemisphere LEMDE domes welded to a four-inch cylindrical center section. The tanks are sized for the 1975/77 Mission. The tanks are mounted to circular skirts which are supported by the beams and the side of the shell. The two 34-inch diameter helium spherical tanks are each mounted from trusses atop a propellant tank. The four 16-inch diameter spherical nitrogen tanks for the Attitude Control System are mounted in a balanced fashion from the beams.

The equipment module is a circular 120-inch diameter shell 59 inches high. The structure is a semimonocoque with 16 stringers to accommodate the mounting provisions for the 16 electronic packages. The skin shear load is carried around the skin cutouts, in the area of the packages, by upper and lower ribs and by using the electronic package outer radiating surface as a shear web. This cutout provides good radiative thermal transfer between the back of each package and the interior of the Spacecraft. Excellent accessibility is provided to the cabling which is located in harness trays above and below the packages, external to the shell. The electronic module is easily installed to the propulsion module by bolts attaching through external mating rib flanges.

The emergency Capsule/Spacecraft separation ring is attached to the top of the electronic module. The emergency separation can be accomplished by firing an MDF device which is attached to the separation ring. The bolts which attach the separation ring to the Spacecraft and the bolts which attach the Capsule to the separation ring are installed through external mating flanges.

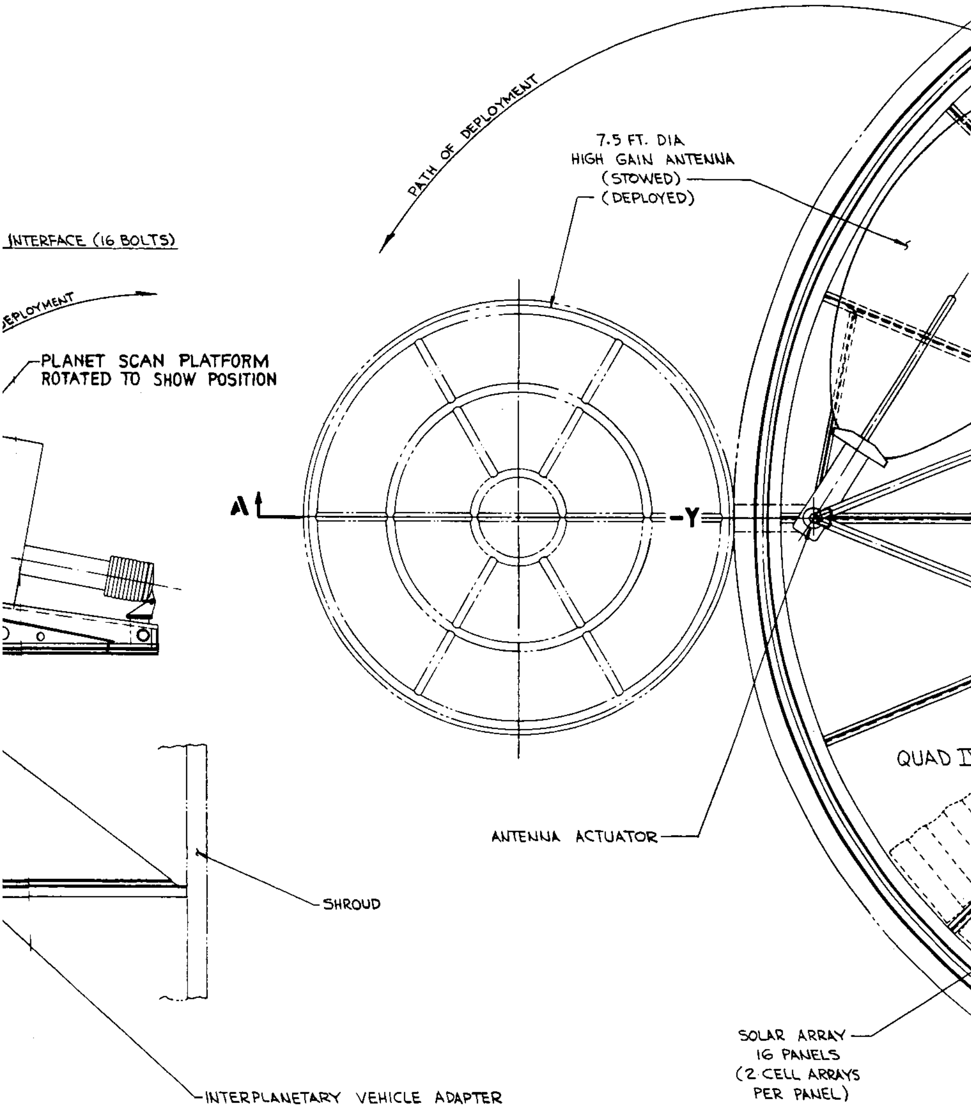
The antennas, planet scanner, and other sensor mounting and deployment provisions are incorporated in the solar array ribs.

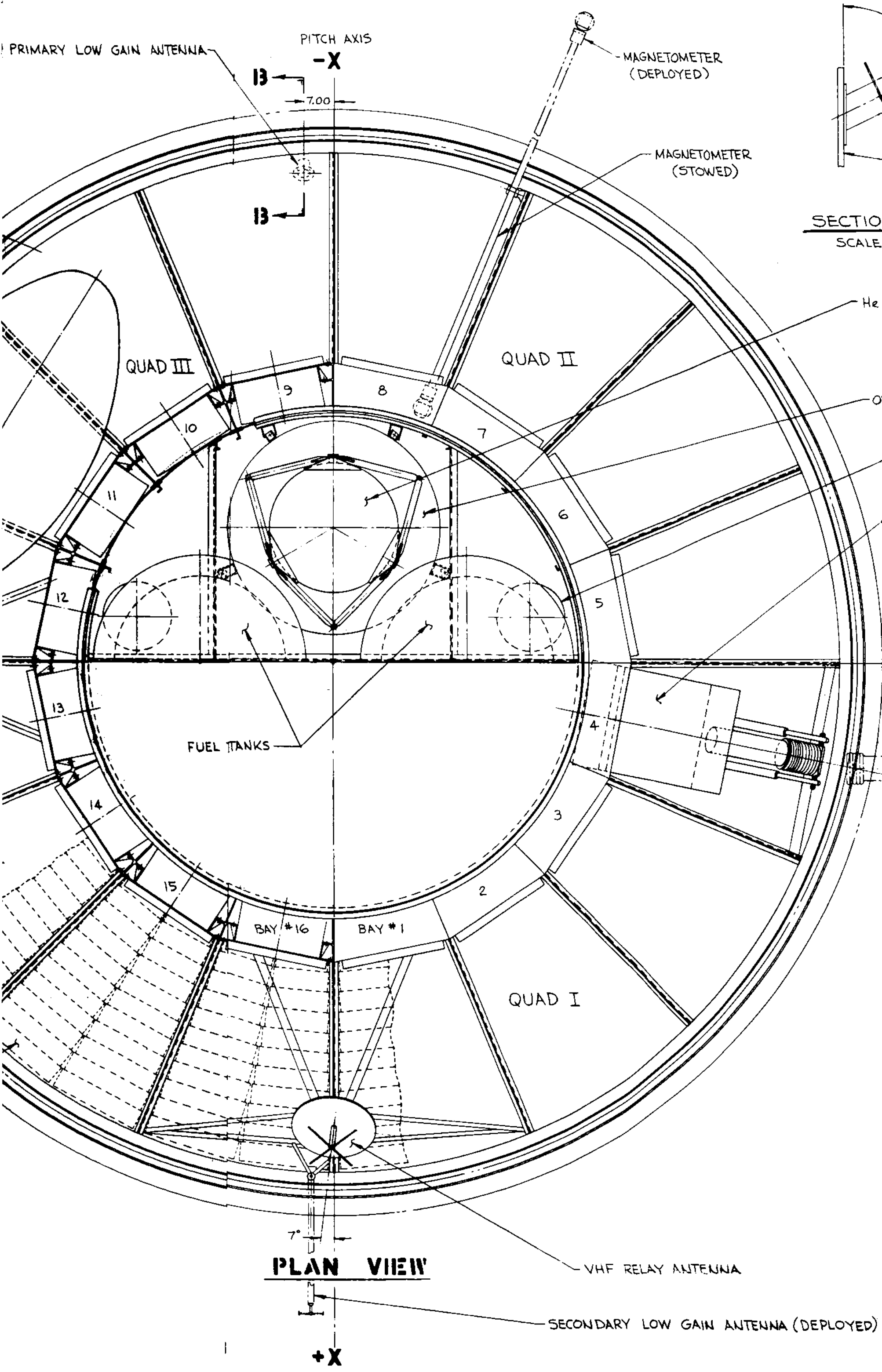

STA. 83.0
STA. 24.0
HIGH GAIN ANT. (DEPLOYED)
STA. 0.0
STA - 49.0


HERMAL CONTROL LOUVER ASSY
ELECTRONIC EQUIPMENT ASSY (16 PLACES)
+Z
ROLL AXIS
FUEL TANK
FLIGHT CAPSUL
PATH OF
SOLAR CELLS ON THIS SURFACE
ENGINE GIMBAL ACTUATOR
SEPARATION SYSTEM (MDF)
MICROMETEOROID BARRIER
LAUNCH VEHICLE INTERFACE
ENGINE GIMBAL LIMIT
6°
RETRO PROPULSION
-Z

SECTION A-A

2


PATH OF DEPLOYMENT
7.5 FT. DIA
HIGH GAIN ANTENNA
(STOWED)
(DEPLOYED)
INTERFACE (16 BOLTS)
DEPLOYMENT
PLANET SCAN PLATFORM
ROTATED TO SHOW POSITION
A
-Y
QUAD I
ANTENNA ACTUATOR
SHROUD
SOLAR ARRAY
16 PANELS
(2 CELL ARRAYS
PER PANEL)
INTERPLANETARY VEHICLE ADAPTER


PRIMARY LOW GAIN ANTENNA
PITCH AXIS
-X
B
7.00
B
MAGNETOMETER (DEPLOYED)
MAGNETOMETER (STOWED)
SECTIO
SCALE
He
QUAD III
QUAD II
QUAD I
9
8
10
7
11
6
12
5
13
4
14
3
15
2
BAY #16
BAY #1
FUEL TANKS
7°
VHF RELAY ANTENNA
SECONDARY LOW GAIN ANTENNA (DEPLOYED)
+X

PLAN VIEW

4

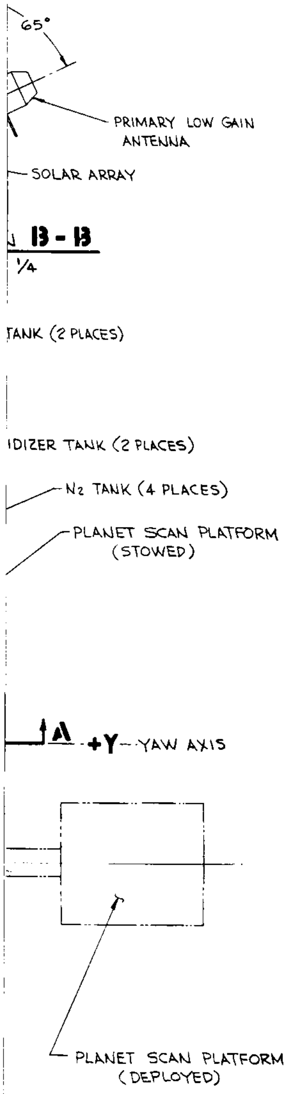


Figure 3.3-5. Modified LEMDE Configuration

Spacecraft meteoroid shielding and thermal control are provided by a four-piece conical assembly attached to the bottom of the propulsion module, and at the top of the Spacecraft by a flat disc attached to the separation ring. Temperature control at the sides is provided by a thermal blanket attached externally to the skin, which provides adequate meteoroid shielding.

4.0 PROPULSION. This section contains a description of the various propulsion systems considered in arriving at the preferred design. Primary effort was expended in examining the LEM Descent Stage, Titan III Transtage, and solid rocket systems, as specified in the work statement.

In the case of the solid propellant retropropulsion system, three basic motor designs were investigated. A description of these systems and the rationale for selection of the modified Minuteman second stage is presented in Section 4.1. Both monopropellant and bipropellant Midcourse/Orbit Adjust (MC/OA) systems were considered for use with the solid retropropulsion system. The reasons for choice of the monopropellant are discussed in Section 4.2.

For the LEMDE and Transtage systems, necessary modifications to adapt these stages to VOYAGER were investigated. In addition, modifications that could be made to these systems to improve the overall VOYAGER design were studied.

Finally, two alternate concepts that could prove advantageous to VOYAGER were investigated and are presented in Section 4.5. These are a four, 2200-pound thrust chamber arrangement and a beryllium thrust chamber design.

4.1 Solid Propellant Retropropulsion. Orbit insertion solid propellant motor designs developed for the 1971/73 VOYAGER Missions and the 1975/77 Missions comply with system requirements established by the Jet Propulsion Laboratory (JPL). This section of the report describes typical motor configurations developed to provide the basis for parametric studies conducted by General Electric.

4.1.1 Candidate System Descriptions

4.1.1.1 1971/73 Missions

4.1.1.1.1 Requirements

a. Velocity Increment - Not less than 2.0 km/sec, with a design goal of 2.2 km/sec.

b. Payload Weight - 5500 pounds plus weight of the interplanetary trajectory correction and Mars orbit-trim propulsion system. Assume that 100% of the propellant allotted for the midcourse trajectory correction has been used at the time of orbit insertion.

c. Payload Acceleration - Shall not be greater than 3.0g at any time during orbit-injection motor firing.

d. Motor Envelope - The motor shall fit a nominal envelope 208 inches long by 100 inches in diameter. Effort shall be made to minimize length within the envelope.

4.1.1.1.2 Aerojet Modified Minuteman

A. Overall Description - The candidate 1971/73 Missions motor, shown in Figure 4.1-1, is a modified second stage Minuteman Wing VI motor and is described fully in VC238FD102, Volume A. The significant differences between the proposed motor and the Minuteman motor

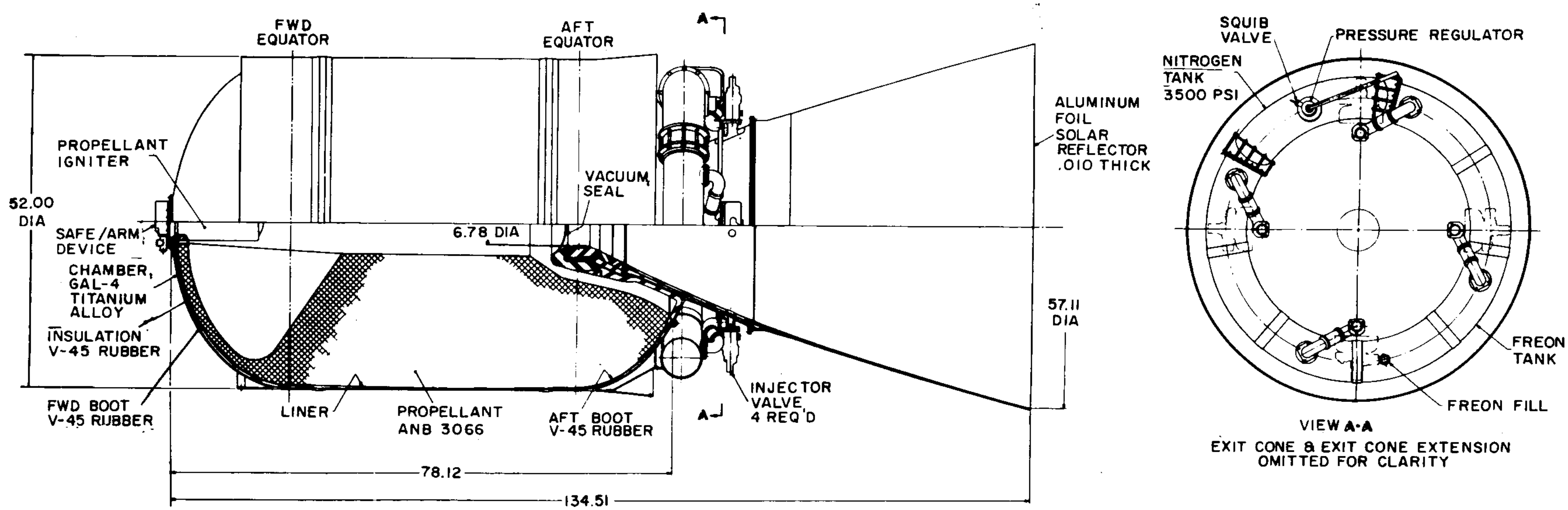


Figure 4.1-1. Solid Propellant Orbit Injection Motor

(shown in Figure 4.1-2 of the Classified Supplement) are a reduced motor length, smaller throat diameter, and a larger nozzle expansion ratio. Minuteman performance parameters are presented in NOTE 4 of the Classified Supplement while performance of the modified Minuteman has been presented in VC238FD102, Volume A.

The second-stage Minuteman Wing VI motor chamber is fabricated of 6Al-4V titanium alloy and is insulated internally with premolded silica-loaded butadiene acrylonitrile rubber (Gen-Gard V-45). The propellant, designated ANB-3066, is a carboxy-terminated polybutadiene formulation with 88% solids. The propellant grain has a finocyl configuration and is bonded to the motor case except in the forward and aft head areas where the propellant surface is restricted from burning at ignition by "boots" (a thin layer of Gen-Gard V-45 insulation material). The propellant is bonded to the insulation and boots with a polybutadiene liner material. The nozzle is a single submerged contoured nozzle with an exit cone expansion ratio of 24.8:1. Motor ignition is provided by a solid propellant igniter; the propellant is identical to that used in the motor. Flight control is achieved with an independent liquid-injection thrust vector control (TVC) subsystem and a hot gas roll control (RC) subsystem.

B. Motor Sizing Studies - Motor sizing studies were performed for the 1971/73 Missions in accordance with the above requirements. The specific values of velocity increment and payload are listed in Table 4.1-1.

TABLE 4.1-1. VELOCITY INCREMENT AND PAYLOAD

	ΔV (km/sec)	Payload (lbm)
Monopropellant MC/OA		
Case I	2.2	6280
Case II	2.0	6202
Bipropellant MC/OA		
Case III	2.2	6114
Case IV	2.0	6075

The motor sizing studies resulted in a series of plots shown in Figures 4.1-3 through 4.1-11 covering the range of payloads and velocity increments of interest. Motor total impulse is the common parameter for the series of plots. These plots represent fully-loaded motor designs; the design point motor (Case I) is indicated on each plot. The relationship between total impulse and payload weight for several velocity increments is shown on Figure 4.1-3. Total impulse is governed by propellant weight and expansion ratio. Propellant weight is varied by lengthening or shortening the barrel section of the Minuteman Wing VI motor, therefore each point on Figures 4.1-3 through 4.1-11 represents a fully-loaded motor.

Previous studies have shown that for the proposed exit cone design, the tradeoff point where increased expansion cone weight exceeds decreased propellant weight occurs at expansion ratios above 100:1. However, above a 70:1 expansion ratio, the savings in total motor weight is small. Therefore to keep over-all motor length low, consistent with efficient motor design, an expansion ratio of 70:1 was selected. Propellant weight and total motor weight are plotted versus total impulse for different expansion ratios on Figures 4.1-4 and 4.1-5, respectively. The effective specific impulse and effective mass fraction versus total impulse are plotted on Figures 4.1-6 and 4.1-7, respectively. Effective values of specific impulse

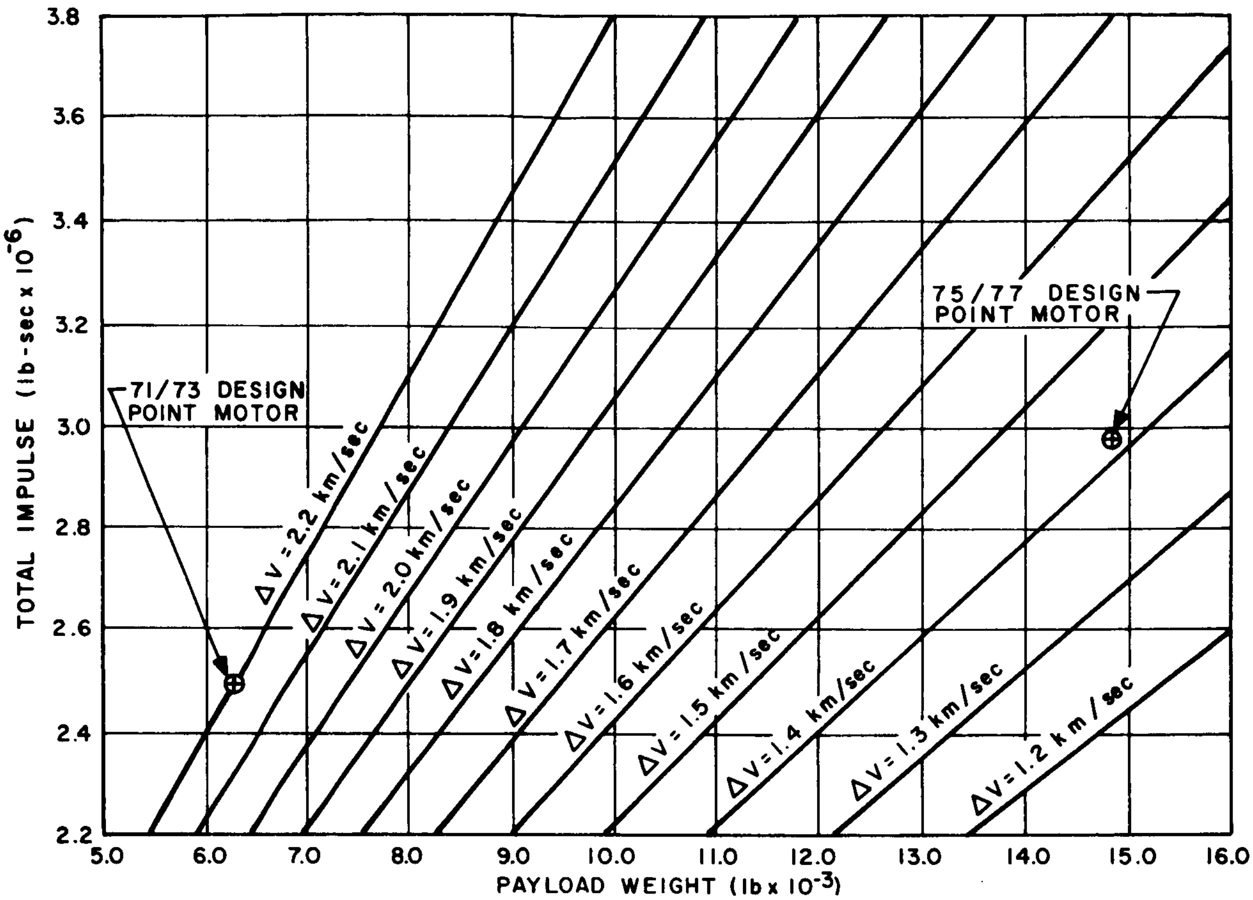


Figure 4.1-3. Total Impulse Versus Payload Weight-Modified Minuteman Motor (ANB-3066 Propellant)

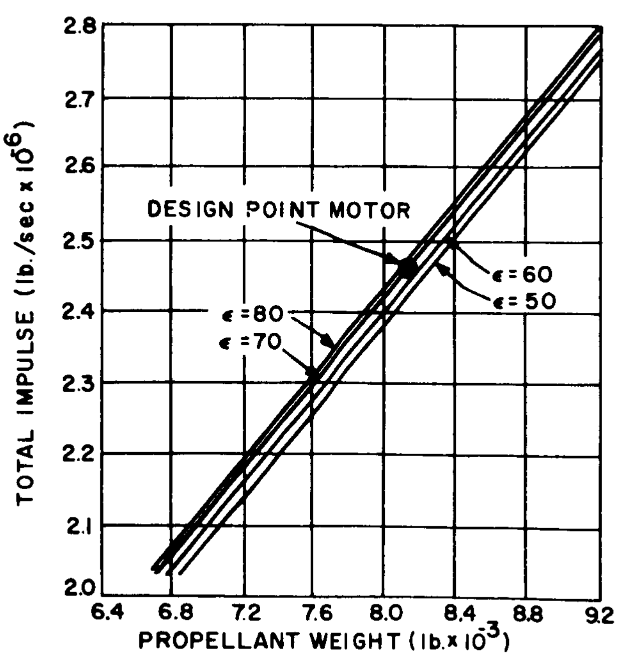


Figure 4.1-4. Total Impulse Versus Propellant Weight-Modified Minuteman Motor

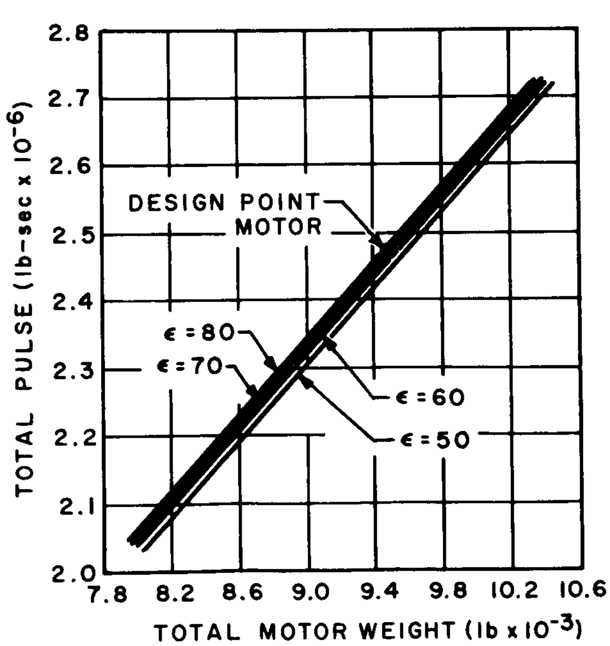


Figure 4.1-5. Total Impulse Versus Motor Weight-Modified Minuteman Motor

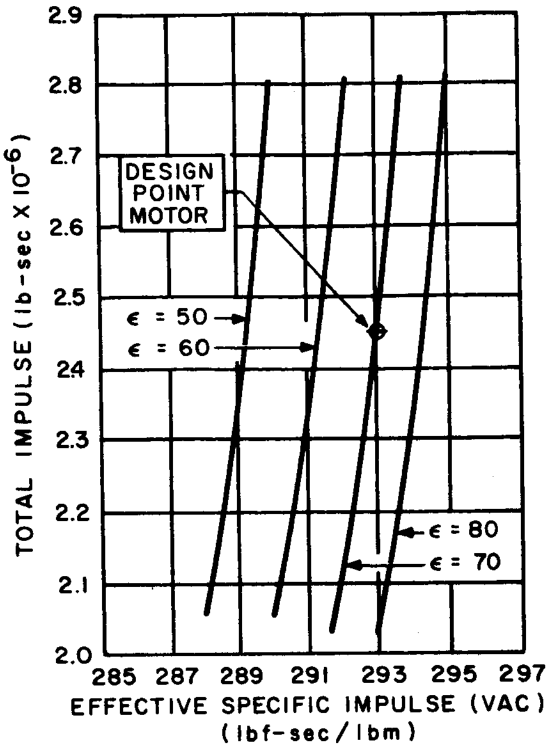


Figure 4.1-6. Total Impulse Versus Effective Specific Impulse-Modified Minuteman Motor

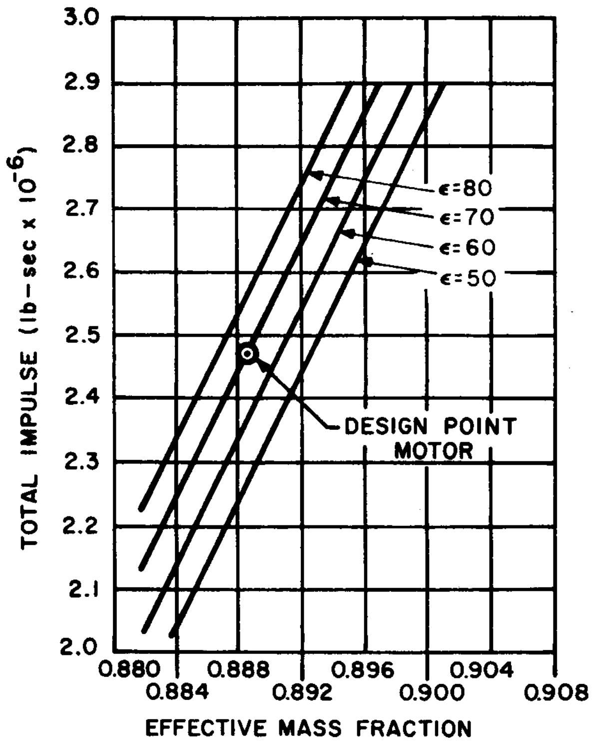


Figure 4.1-7. Total Impulse Versus Effective Mass Fraction-Modified Minuteman Motor

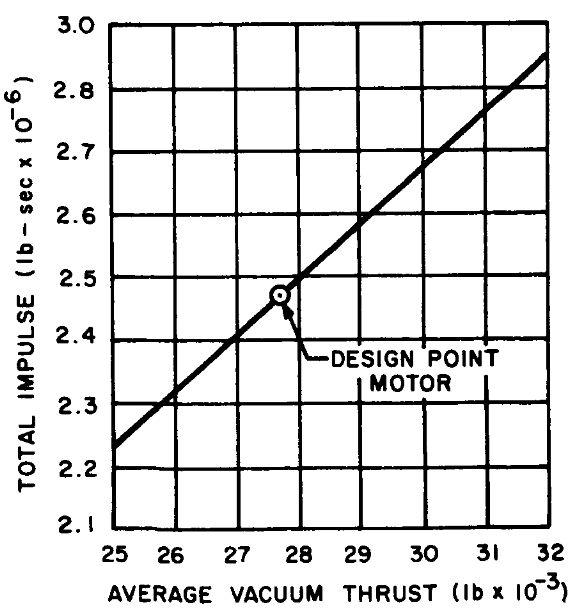


Figure 4.1-8. Total Impulse Versus Average Vacuum Thrust-Modified Minuteman Motor

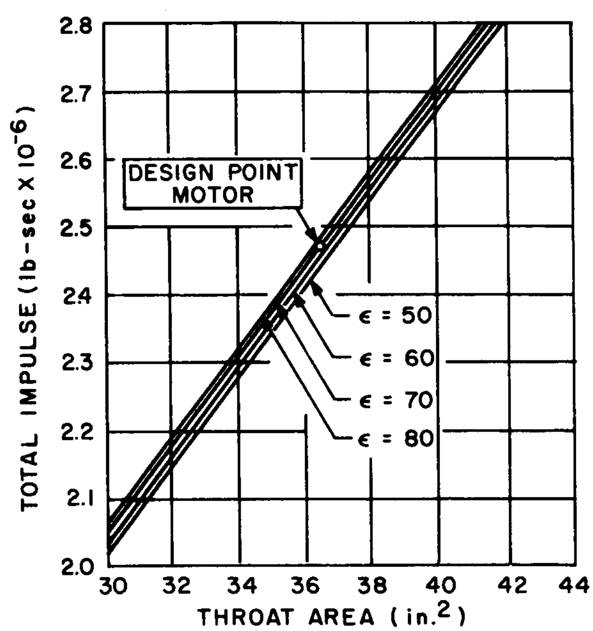


Figure 4.1-9. Total Impulse Versus Throat Area-Modified Minuteman Motor

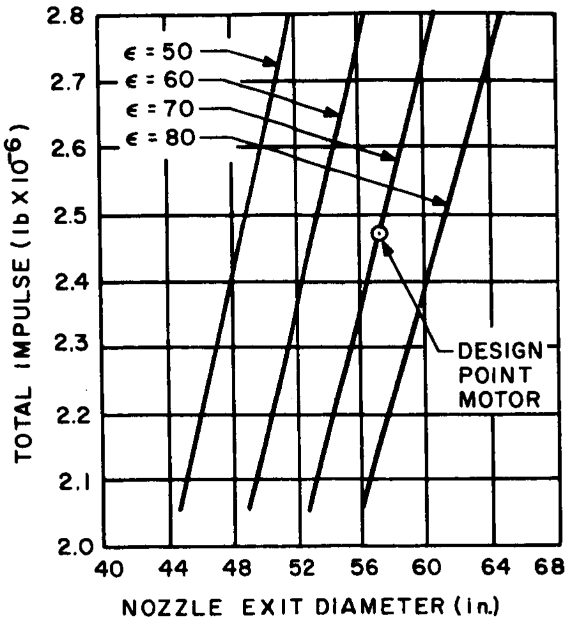


Figure 4.1-10. Total Impulse Versus Nozzle Exit Diameter-Modified Minuteman Motor

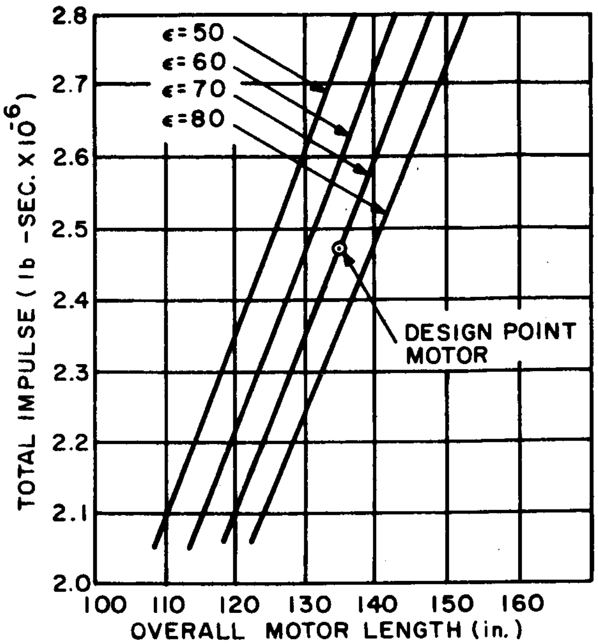


Figure 4.1-11. Total Impulse Versus Overall Motor Length-Modified Minuteman Motor

and mass fraction are given since insulation materials and TVC fluid are consumed during the firing. The effective specific impulse is found by the following relationship:

$$I_{s_{eff}} = \frac{(W_p)(\text{Propellant } I_s)+(W_{\text{Expended Inerts}})(\text{Expended Inerts } I_s)+(W_{\text{TVC Fluid}})(\text{TVC}_{\text{Fluid}}\, I_s)}{W_p + W_{\text{Expended Inerts}} + W_{\text{TVC Fluid}}}$$

The effective mass fraction is found as follows:

$$\text{Mass Fraction, Effective} = \frac{W_p + W_{\text{Expended Inerts}} + W_{\text{TVC Fluid}}}{W_p + W_{\text{Expended Inerts}} + W_{\text{TVC Fluid}} + W_{\text{Unexpended Inerts}}}$$

Motor average vacuum thrust versus total impulse is shown on Figure 4.1-8. Shown on Figure 4.1-9 is the throat area versus total impulse. Nozzle exit diameter and overall motor length versus total impulse for various expansion ratios are given in Figures 4.1-10 and 4.1-11, respectively.

An additional design requirement is that the same component and hardware be used for all 1971/73 Mission concepts studied. Thus, based onthe above sizing studies, the prime motor is sized for the largest velocity increment (2.2 km/sec) and payload (6280 lb) and is designated Case I. The alternate motors (Cases II, III, and IV defined above) will require less propellant to meet mission objectives. The prime motor can be readily adapted to meet a wide range of mission velocity and payload requirements by using less propellant. The ballistic curves of the off-loaded motor will be identical to the curves of the fully-loaded motor, except that ignition time on the off-loaded motor will correspond to some point after ignition on the fully-loaded motor. This is accomplished by increasing the thickness of the casting core to correspond to a propellant burn-back line, which results in the desired propellant weight to achieve the required velocity increment. Chamber, nozzle, insulation, and TVC will be identical to the fully-loaded motor; only the grain configuration is different. A motor can be tailored to a specific velocity increment and payload weight by this method two to three months prior to the required delivery date.

To compensate for small changes in the required velocity increment or payload weight, last minute adjustments could be made in the orbit-injection motor total impulse by varying the nozzle expansion ratio. This is accomplished by shortening the nozzle exit cone extension. If the exit cone extension were to be removed entirely, the velocity increment would be reduced from 2.2 to approximately 2.1 km/sec for a 6280-pound payload. Since shortening the nozzle will increase heat flux to the Spacecraft, this method will have to be investigated further in Phase IB to determine if it is feasible.

A plot of motor weight versus payload for velocity increments of 2.2 and 2.0 km/sec is shown in Figure 4.1-12. This plot presents the relationship between payload and velocity increment as propellant is off-loaded from the basic motor. Motor inerts and external configuration are held constant; only propellant weight is varied.

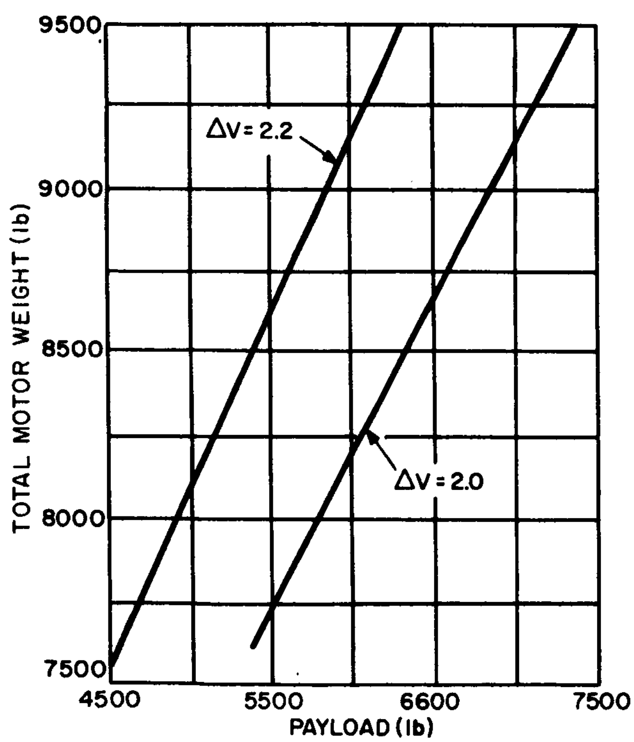


Figure 4.1-12. Total Motor Weight Versus Payload-Modified Minuteman Motor

4.1.1.1.3 Aerojet Ovaloid

A. Overall Description - As an alternative to the modified Minuteman motor described above, a glass-filament chamber motor, shown in Figure 4.1-13, has been considered. The motor has an ovaloid glass-filament, epoxy-resin chamber with a major diameter of 74 inches and an overall motor length of 84 inches. The major components are: aluminum polybutadiene propellant (ANB 3066); a finocyl grain configuration; General Tire and Rubber Company's Gen-Gard V-45 internal insulation (a silica-loaded nitrile-rubber); a glass filament ovaloid chamber; a single submerged contoured nozzle with a tungsten throat and plastic exit cone; and a squib-initiated, controlled pressure igniter with a safe and arm device.

Thrust vector control is achieved with a slightly modified Minuteman liquid injection TVC system.

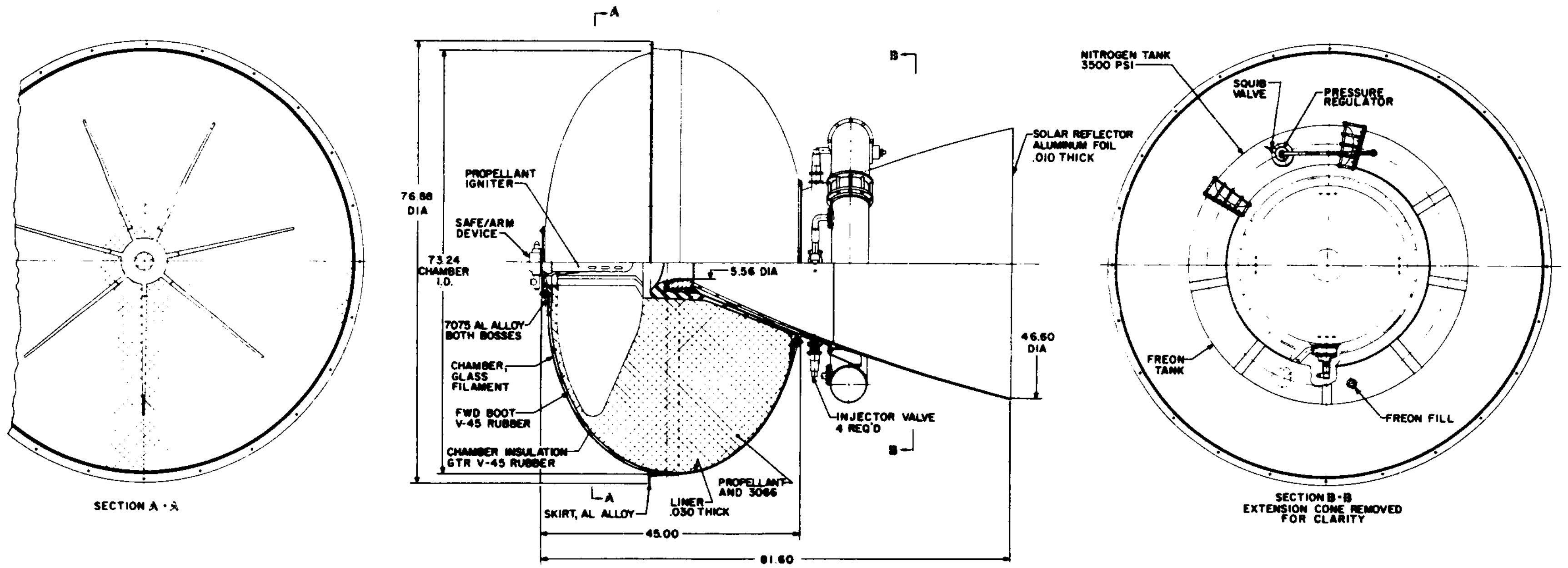


Figure 4.1-13. Solid Propellant Orbit Injection Motor with Glass Filament Chamber

The primary differences between the ovaloid and the modified Minuteman motor is the use of the glass-filament chamber to reduce weight, and a larger case diameter to reduce motor length. With the exception of the glass chamber, the propellant and other materials are the same as those used on the modified Minuteman motor. For most components, such as the TVC system, ignition system, and nozzle, the same basic design concepts are retained. The motor design assures that spacecraft heating will be held within acceptable limits. Insulation thicknesses are based on a 50°F rise in case temperature during firing. An ablative plastic exit cone is used to minimize heat emission from the exterior of the exit cone. The submerged nozzle prevents the hot throat insert from radiating heat directly to the Spacecraft during the postfire heat-soak period.

B. Motor Sizing Studies - Motor sizing studies were performed for the 1971/73 Mission in accordance with the requirements defined for the modified Minuteman. Subject to these requirements, an analysis was conducted to determine the optimum operating characteristics of a motor (designated Case I) sized for the largest velocity increment (2.2 km/sec) and payload (6212 pounds) established.

The results of the optimization study are plotted in Figure 4.1-14 and show average chamber pressure versus total motor weight for various expansion ratios. Minimum motor weight for each expansion ratio is defined by a curve showing the pressure at which this occurs. Curves of constant motor lengths show the effect on motor weight due to length restrictions, while curves of constant accelerations show the effect of limitations in axial acceleration. At the intersection of a particular set of motor length and acceleration curves, the average motor operating pressure, nozzle expansion ratio and motor weight are defined.

An additional design requirement is that the same components and hardware be used for all the 1971/73 Mission concepts studied. The alternate motors (Cases II, III and IV) will require less propellant to meet mission objectives. The prime motor can be readily adapted to meet a wide range of mission velocity and payload requirements by using less propellant. The ballistic curves of the off-loaded motor will be identical to the curves of the fully loaded motor, except that ignition time of the off-loaded motor will correspond to some point after ignition of the fully loaded motor. This is accomplished by increasing the thickness of the casting core to correspond to a propellant burn-back line which, in turn, results in the desired propellant weight to achieve the required velocity increment. Chamber, nozzle, insulation, and TVC designs will be identical to those of the fully-loaded motor; only the grain configuration is different. A motor can be tailored to a specific velocity increment and payload weight by this method as late as two or three months before the required delivery date.

To compensate for small changes in the required velocity increment or payload weight, last minute adjustments could be made in the orbit-injection total impulse by varying the nozzle expansion ratio. This is accomplished by shortening the nozzle exit cone extension. If the exit cone extension were to be removed entirely, the velocity increment would be reduced from 2.2 to approximately 2.1 km/sec for a 6212-pound payload.

A plot of motor weight versus payload for velocity increments of 2.2 and 2.0 km/sec is shown on Figure 4.1-15. This plot shows the relationship between payload and velocity increment as propellant is off-loaded from the basic motor. Motor inerts and external configuration are held constant; only propellant weight is varied.

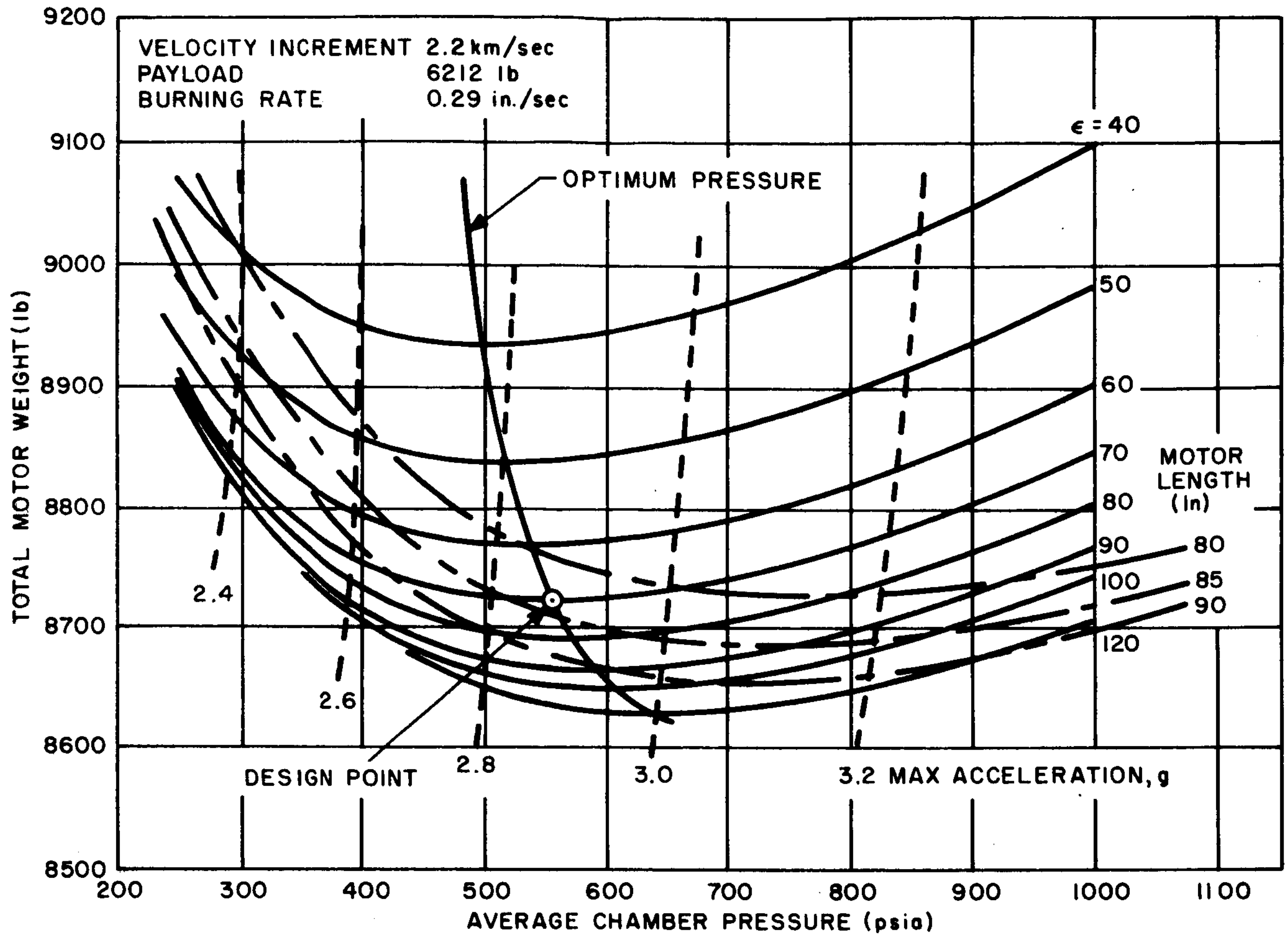


Figure 4.1-14. Parametric Data-New Motor

4.1.1.1.4 Thiokol TU-533 Series

A. Overall Description - This study included the preliminary design of several motor configurations. For the 1971/73 VOYAGER Mission, a total of seven configurations were considered. Two preliminary configurations, the TU-533B and TU-533D motors, were designed to impart velocity increments of 2.2 km/sec to Spacecraft weighing 6227 and 6096 pounds, respectively, not including the orbit insertion motor weights. Two additional designs which use the same hardware as the TU-533B and TU-533D motors, but contain less propellant, are identified as the TU-533A and TU-533C motors, respectively. These four motor designs were not completely optimized as they are sufficiently close to the optimized configurations to reflect the correct trends.

The propulsion subsystems studied were solid propellant rocket motors equipped with Freon 114B2 secondary injection for thrust vector control. Each motor consists of a solid propellant grain, an insulated fiberglass case, an ablative nozzle with a refractory (Graphite 90) throat insert, and a Pyrogen igniter with a safe/arm device. The LITVC system for each

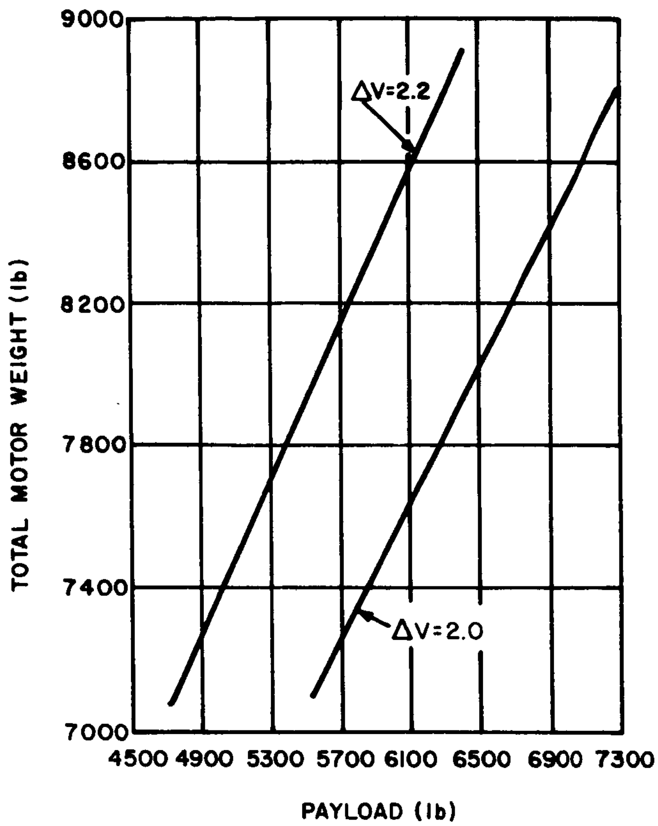


Figure 4.1-15. Total Motor Weight Versus Payload-New Motor

motor consists of four modulating injector valves hydraulically operated in pairs by two electrically controlled servovalves, an injectant tankage and pressurization system, and sufficient Freon 114B2 for the intended mission. The materials used are compatible with the space environment and mission duration.

The propellant is a high performance formulation using a carboxyl terminated polybutadiene (HC series) binder (14% by weight), ammonium perchlorate oxidizer (70% by weight), and aluminum fuel (16% by weight). This propellant will deliver a vacuum specific impulse of 291.2 lbf-sec/lbm at an expansion ratio of 50:1 in motors of the VOYAGER size. The 52-inch diameter motor case is fabricated with S994 fiberglass preimpregnated with U.S. Polymeric E-717 epoxy resin. The internal insulation is a layer of silica filled nitrile butadiene rubber (NBR) used as a case bladder and silica and asbestos filled NBR in areas of long exposure to the combustion products.

B. Motor Sizing Studies

1. Motor for Use with Monopropellant MC/OA System - The solid propellant propulsion system identified as the TU-533B motor was sized to be compatible with the monopropellant midcourse and orbit adjust propulsion system. This motor, designed for a payload of 6227 pounds and a velocity increment of 2.2 km/sec, was the optimized version. Since the TU-533B motor configuration was identified prior to completion of the optimization studies, it was redesigned and designated TU-533E. The design differences are:

a. The nozzle expansion ratio is 35:1 instead of 39:1. Preliminary optimization studies indicated that minimum motor weight would be provided with this value.

b. The average chamber pressure is 700 psia instead of 500 psia. Initial studies indicated that thrust regressivity requirements would dictate a pressure level of this magnitude to assure that chamber pressure near motor burnout is not significantly lower than 300 psi as required for efficient motor operation.

c. The thrust vector control system will provide a total side impulse of approximately 1.4% instead of approximately 1.0%.

d. The nozzle divergent and turnback angles of the typical nozzle are 25 and 10 degrees, respectively, as compared to 22.8 and 12 degrees, respectively for the optimized nozzle. These selections were based on the configuration of the Surveyor nozzle.

These design parameter differences result in an increased motor weight of 423 pounds. Approximately 275 pounds of this increase can be attributed to the TVC requirements.

The TU-533B motor has a total weight of 9559 pounds and an overall (tip to tip) length of 111.7 inches, while the TU-533E motor weighs 9136 pounds. An inboard profile of this motor is shown in Figure 4.1-16.

A nominal velocity increment of 2.2 km/sec, assuming loss of inert weight (insulation and Freon) at a constant rate during motor operation and a maximum acceleration of 3.0g, will be imparted to a payload of 6227 pounds during 82 seconds of motor operation at 60°F. The propulsion subsystem effective mass fraction (total motor weight loss divided by motor ignition weight) is 0.919 for the TU-533B and 0.918 for the TU-533E. The effective delivered specific impulse (total motor impulse divided by total motor weight loss) is 275.6 lbf-sec/lbm for the TU-533B and 282.4 for the TU-533E.

The off-loaded version of this motor, TU-533A (Figure 4.1-17) has the same overall dimensions as the basic motor. It has a total weight of 8460 pounds, an effective mass fraction of 0.908, and an effective specific impulse of 275.8.

2. Motor for use with Bipropellant MC/OA System -- The motor designed for compatibility with the bipropellant MC/OA propulsion system (identified as the TU-533D, Figure 4.1-18) is similar to the design of the TU-533B motor. The major differences are the overall performance requirements of the motor. The proposed TU-533D has a total weight of 9347 pounds and an overall length of 107.4 inches (excluding S&A).

The TU-533D motor will impart a velocity increment of 2.2 km/sec and a maximum acceleration of 3.0g to a payload weight of 6096 pounds during 82 seconds of motor operation at 60°F. The propulsion sybsystem effective mass fraction is 0.917. The effective delivered specific impulse is 275.6 lbf-sec/lbm.

The off-loaded version of this motor, TU-533C (Figure 4.1-19) has the same overall dimensions as the basic motor. It has a total weight of 8307 pounds, an effective mass fraction of 0.906, and an effective specific impulse of 275.7 seconds.

4.1.1.2 1975 and 1977 Missions

4.1.1.2.1 Requirements

a. Velocity increment - maximum possible, consistent with a total propulsion system weight of 15,000 pounds including the orbit-injection motor, interplanetary trajectory correction, and Mars orbit-trim propulsion system.

b. Payload - 13,500 pounds plus the weight of the interplanetary trajectory correction and Mars orbit-trim propulsion system.

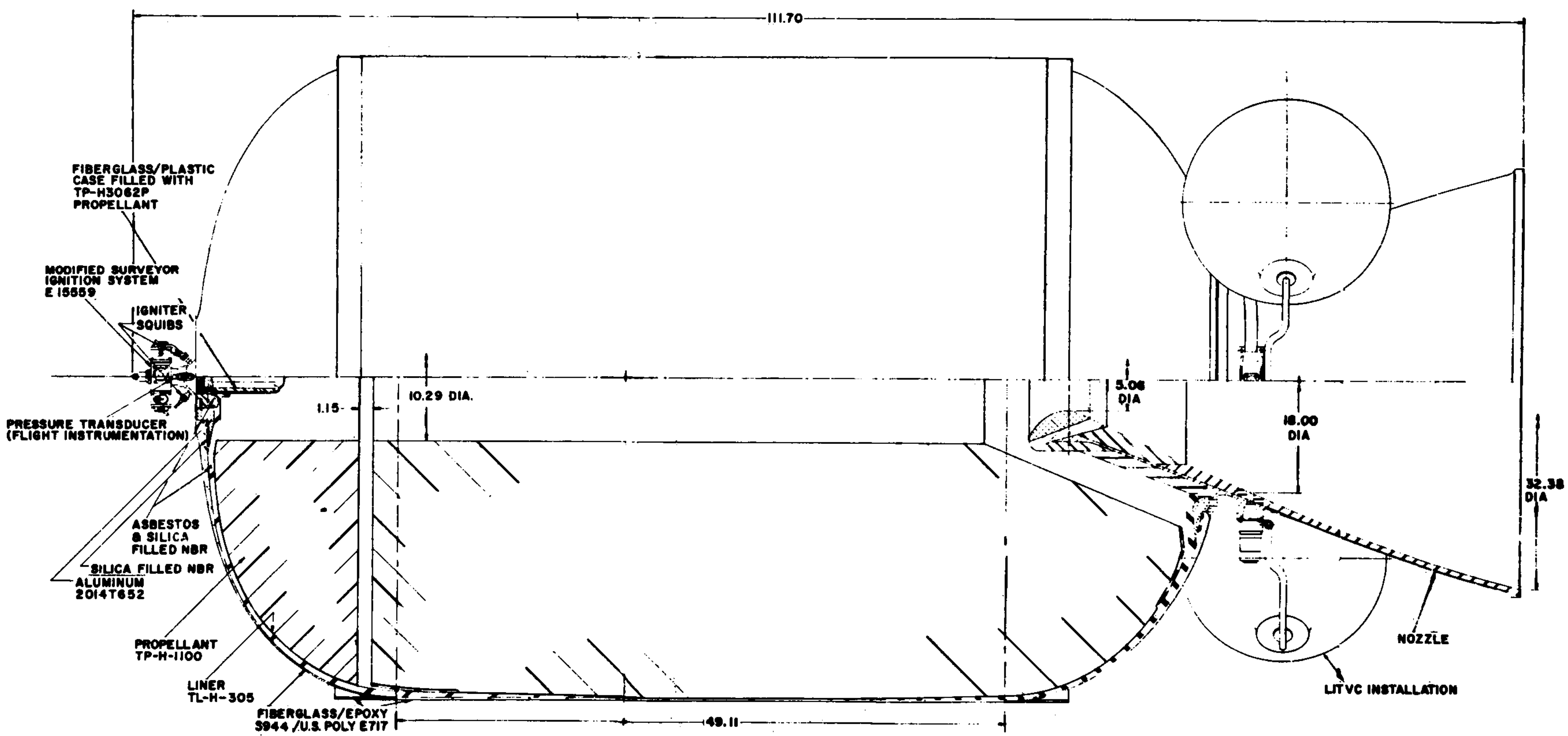


Figure 4.1-16. TU-533B Motor Assembly

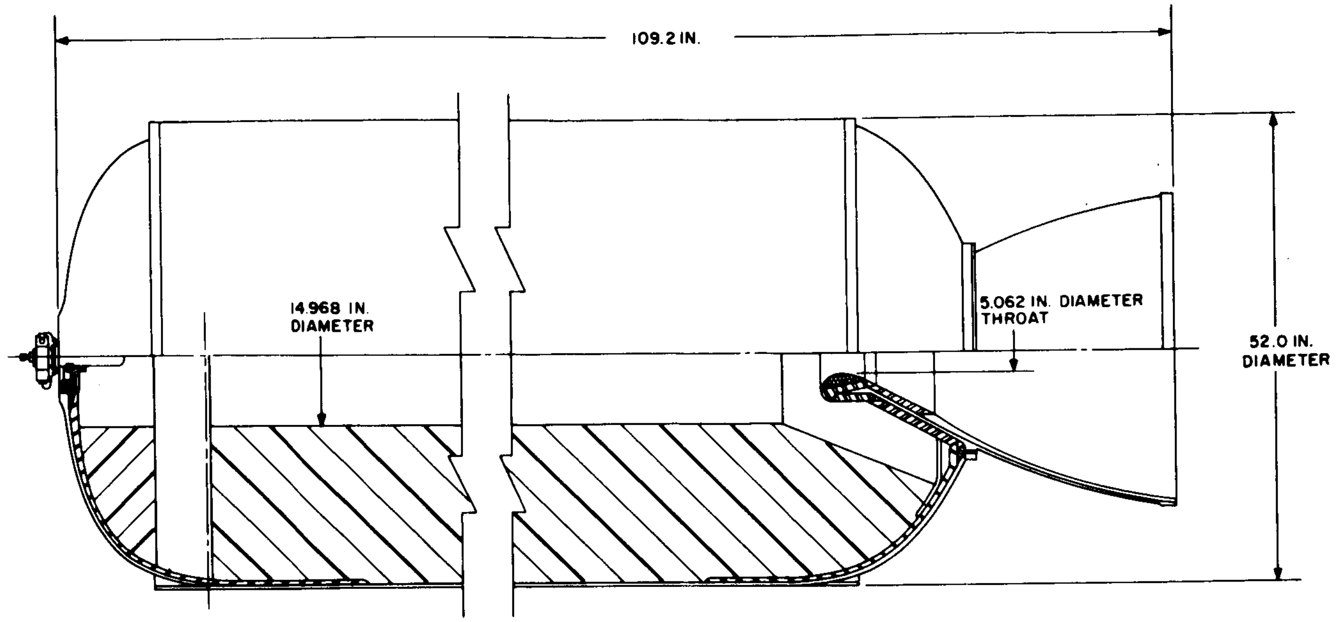


Figure 4.1-17. TU-533A Motor Assembly

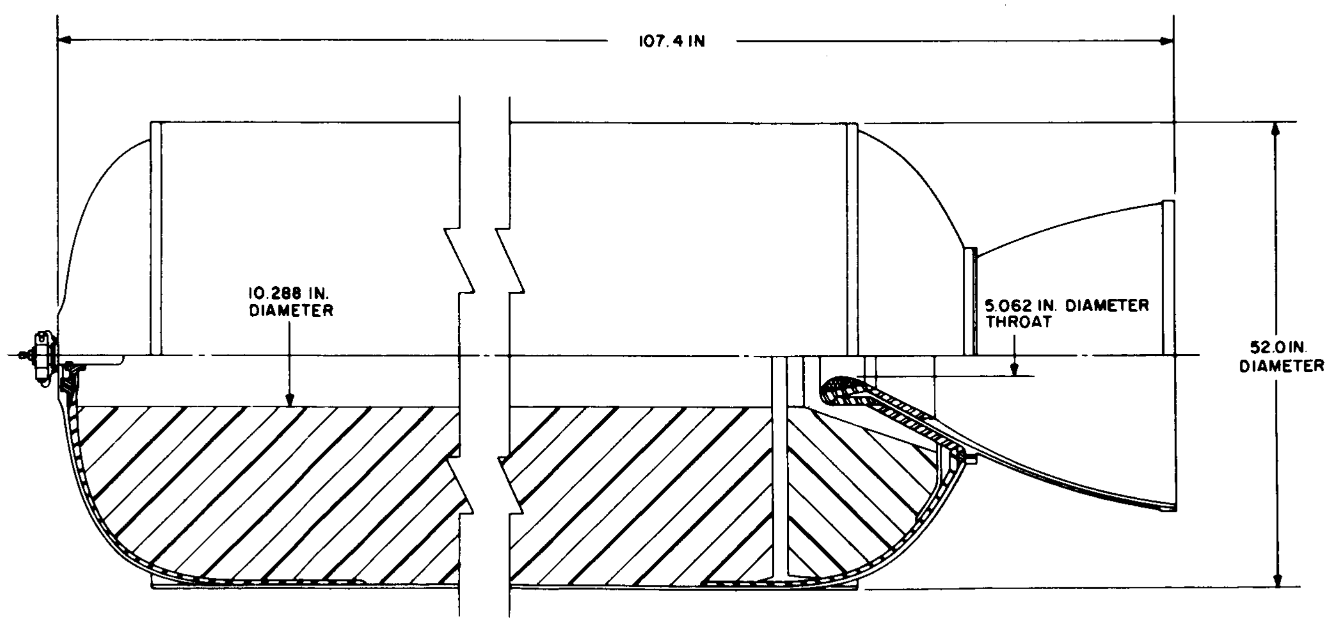


Figure 4.1-18. TU-533D Motor Assembly

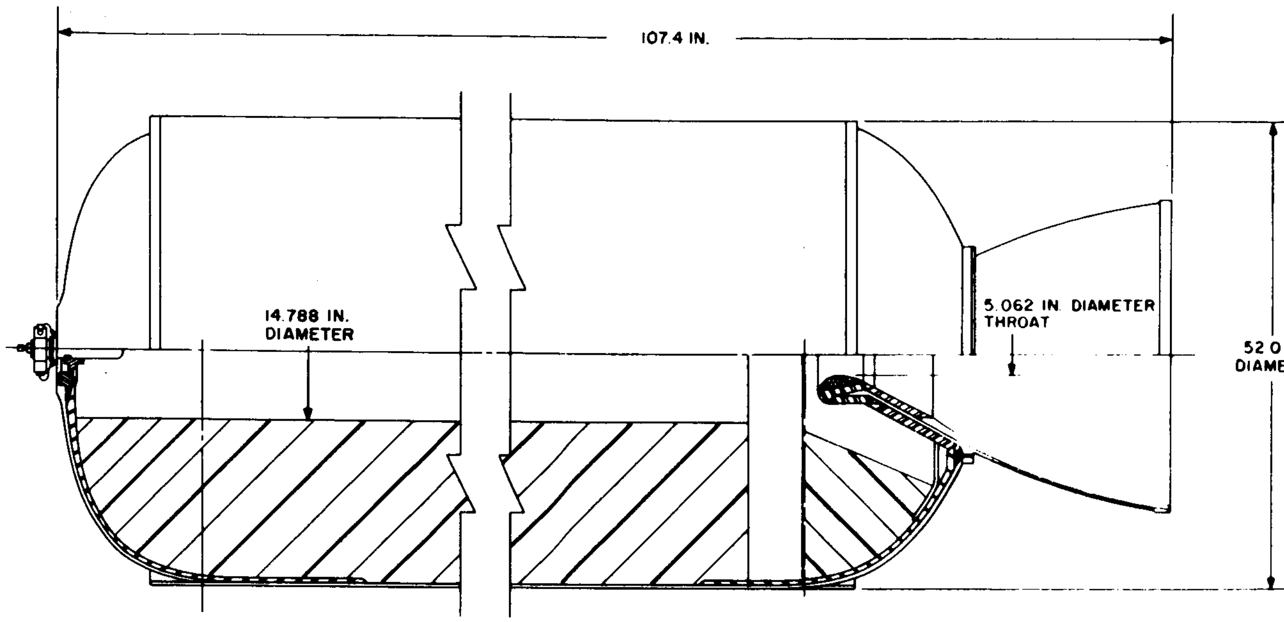


Figure 4. 1-19. TU-533C Motor Assembly

c. Payload acceleration - shall not be greater than 3.0g at any time during orbit-insertion motor firing.

d. Motor envelope - the motor shall fit in a nominal envelope 208 inches long by 100 inches in diameter. Effort shall be made to minimize length within the envelope.

4.1.1.2.2 Aerojet Modified Minuteman

A. Overall Description - The increased propulsion requirements demanded by the 1975/77 Missions will be met by an uprated version of the same basic modified second-stage Minuteman Wing VI motor described for the 1971/73 Missions. The increased requirements can be met by either using more aluminized ANB-3066 propellant or replacing ANB-3066 with ANB-3212, a high energy beryllium propellant. The chamber barrel length will be increased from 45 to 58.4 inches for the increased aluminum propellant and to 63 inches for the beryllium propellant. These motors will be 29.6 and 25 inches, respectively, shorter than the present Minuteman motor. There will be a corresponding increase in throat diameters, from 6.78 inches to 7.50 and 7.67 inches, respectively, to maintain the desired operating pressure. The existing nozzle housing design will still be used. Internal insulation and nozzle insulation thicknesses will be increased to compensate for the increased erosive effects due to beryllium propellant.

The identical TVC system proposed for the 1971/73 Missions will be used. Ample side force capacity exists for the additional total impulse requirements since the present Minuteman Wing VI system is capable of generating side force in excess of that required for the proposed motor.

B. Motor Sizing Studies - A plot of total motor weight with beryllium propellant versus velocity increment for various payload weights is shown in Figure 4.1-20. The same data for the aluminized propellant motor is obtained from Figures 4.1-3 and 4.1-5. These plots can be used to determine the maximum velocity increment obtainable for various payloads consistent with the maximum propulsion system weight of 15,000 pounds.

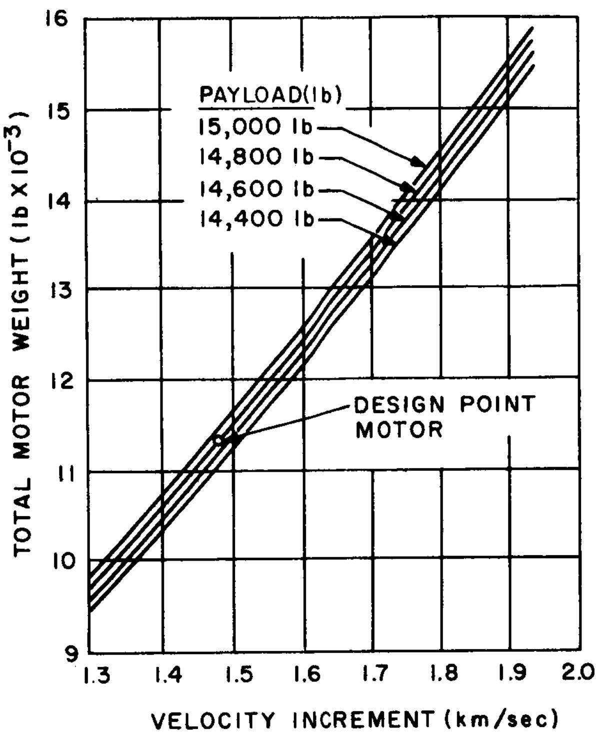


Figure 4.1-20. Total Motor Weight Versus Velocity Increment-Modified Minuteman Motor

C. Performance Characteristics - The beryllium design point motor is capable of applying a 1.485 km/sec velocity increment to a 14,868-pound payload, while the aluminized design point motor applies a 1.418 km/sec velocity increment to the same payload. Tabulated performance and weight data are given in Table 4.1-2 of the Classified Supplement. Maximum acceleration imparted to the payload will be less than 2g for both cases since the payload to thrust ratio is greater than for the 1971/73 motors. The shape of the pressure and thrust versus time curves will be identical to those presented for the 1971/73 motor. Except for differences in length, throat diameter and exit cone diameter, the motor configuration is the same as shown on Figure 4.1-1. Overall beryllium motor length is 161.5 inches, aluminum motor length is 155.7 inches, and the exit cone inside diameter is 64.0 and 62.7 inches, respectively.

D. Component Description - The changes to the motor design discussed in VC238FD102, Volume A, that are required for the 1975/77 Missions, involve the nozzle, motor internal insulation, and propellant. The nozzle ablative insulation and motor internal insulation would be increased if a beryllium propellant was used because of the higher erosive effect. The primary propellant requirements of the orbit-injection motor for the 1975/77 Mission are:

a. Reliability

b. Highest vacuum specific impulse compatible with reliability

c. Propellant burning rates of 0.20 to 0.35 in./sec at 500 psia

d. Mechanical properties equivalent to ANB-3066, the Minuteman Wing VI second-stage propellant proposed in the earlier flight motors.

The high energy propellant under consideration for the 1975/77 Missions is a beryllium-ammonium perchlorate-polybutadiene system, ANB 3212, closely related to the ANB-3066 propellant used in the Stage II Minuteman motors. Impulse data and composition are presented in NOTE 6 of the Classified Supplement.

The burning rate of the current ANB 3212 formulation is 0.25 in./sec at 500 psia with a pressure exponent of 0.3. This burning rate will meet the requirements of current designs. However, the burning rate range for beryllium systems is comparable to that for aluminum systems and is expected to allow the same design versatility.

The ballistic property and mechanical property data are summarized in Table 4.1-3 of the Classified Supplement. Work is underway in the Minuteman Product Improvement program to improve the mechanical properties of the beryllium propellant to match those of the aluminized system. While it is expected that the motor will be effectively sealed, investigations with the aluminum propellant show that removal of potentially volatile components of ANB-3066 to improve vacuum resistence does not harm the basic mechanical properties of the propellant. In addition, it was found that exposure of this propellant to hard vacuum for sixty days at 77°F has little effect on the properties. The shelf-life of the Minuteman propellant exceeds the three-year minimum requirement for Minuteman and accelerated aging data on ANB-3212 indicate that it will be at least comparable. Many of the safety characteristics of ANB-3212 have been determined and an ICC classification of B, nondetonable, has been established. It is expected that the Military Explosive Hazard rating will be Class 2.

4.1.1.2.3 Thiokol TU-535 Series

A. Overall Description - The propulsion systems designed for VOYAGER application in the 1975/77 Missions consider the fact that development will not commence for three or four years after the 1971/73 motor development start. Although the selected approaches must be compatible with a high inherent reliability, the additional time available for materials or component development should be considered.

As for the 1971/73 Mission motors, the proposed orbit insertion propulsion subsystems are solid propellant rocket motors equipped with Freon 114B2 secondary injection for TVC. The major differences in technology proposed for the 1975/77 systems is the use of beryllium fuel in the HC series polymer formulation and the use of higher strength levels of the case fiberglass structure.

The beryllium propellant proposed is a high performance formulation containing 14% HC polymer, 12% beryllium fuel, and 74% ammonium perchlorate oxidizer. This propellant will deliver a vacuum specific impulse of at least 309 lbf-sec/lbm at an expansion ratio of 50:1.

The case diameter for the TU-533B and C motors is 52.0 inches. This series of motors tended to optimize at smaller diameters when length is not considered. Because minimum length is desired, and since interface problems may be alleviated if the 1971/73 and 1975/77 motors have similar diameters, a 52.0-inch diameter was selected for these motors.

B. Motors for use with Monopropellant MC/OA System (TU-535B) - The TU-535B solid propellant propulsion system (Figure 4.1-21) was designed to produce a maximum velocity increment to a 14,886-pound payload with a maximum motor weight of 11,140 pounds. The TU-535B motor will impart a velocity increment of 1.472 km/sec and a maximum acceleration of 2.41g to a payload weight of 14,886 pounds during 71 seconds of motor duration of 60°F. The propulsion subsystem effective mass fraction is 0.921. The effective delivered specific impulse is 299.5 lbf-sec/lbm.

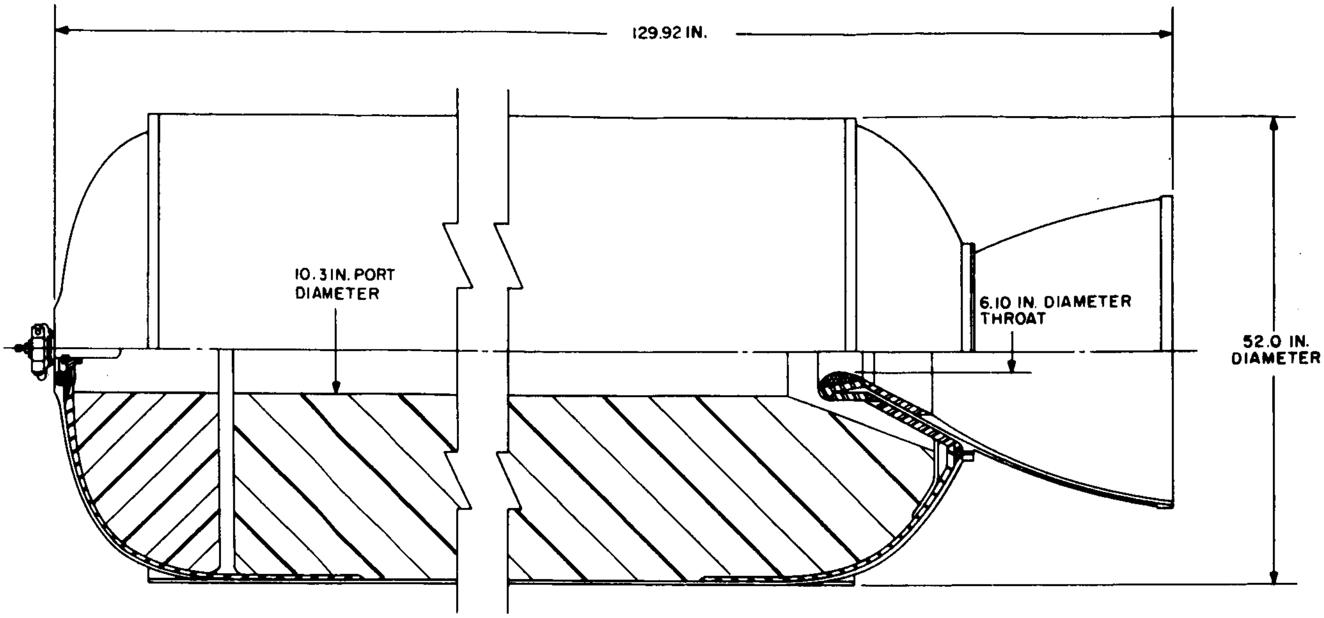


Figure 4.1-21. TU-535B Motor Assembly

C. Motors for use with Bipropellant MC/OA System (TU-535C) - The TU-535C solid propellant propulsion system was designed to produce a maximum velocity increment to a 14,600-pound payload with a maximum motor weight of 12,043 pounds. The TU-535C motor will impart a velocity increment of 1.589 km/sec and a maximum acceleration of 2.67g to a payload weight of 14,600 pounds during 71 seconds of motor operation at 60°F. The propulsion subsystem effective mass fraction is 0.924. The effective delivered specific impulse is 299.8 lbf-sec/lbm.

4.1.2 Roll Moments Induced By Flow Vortexing. A study was conducted by the Wasatch Division of the Thiokol Chemical Corporation in early 1964 regarding the effect of the solid propellant exhaust gases inducing a roll moment by vortexing during expansion through the nozzle. The Division has conducted static firing tests at Arnold Engineering Development Center in an attempt to measure these roll moments. The results of these programs indicated that small roll moments do exist, but of an undetermined magnitude. For this reason, an additional study will have to be conducted during Phase IB to determine the actual magnitude of these roll moments.

Five missile systems were investigated by the Wasatch Division. These were the Sergeant, Scout, Skybolt, Antares II, and Pershing sustainer. The Sergeant, Scout, and Skybolt had propellant compositions containing from 0 to 3% aluminum while the Antares II and Pershing sustainer contained 16 to 18% aluminized propellant. All motors had single nozzles located on the main thrust axis. The results of this study were:

a. Flow vortexing is caused by resonant burning.

b. Resonant burning is a function of propellant grain design, aluminum content, and initial grain temperature.

c. High aluminum content propellants dampen out resonant burning.

d. A 5-point star grain configuration promotes resonant burning, 4- and 6-point star grains attenuate it, and cylindrical perforate grains dampen it.

e. Less resonant burning is associated with low initial grain temperatures. Test points have shown resonant burning at 70° F and 130° F, while there was no resonant burning at minus 30° F.

The Elkton Division of Thiokol static fired four 10,000 lb thrust motors at AEDC. Roll moments of 75, 65, 50, and 50 inch-pounds were measured. However, several oddities occurred during the firings to cause the results to be nonconclusive. These were:

a. The roll moments started at zero and steadily increased to the above values at tail-off.

b. The instrumentation did not return to zero after firing.

c. All roll moments were measured in the same direction.

During flight testing of the Aerojet second stage Minuteman Wing VI motor, missile roll was measured with the roll control system in null position. These measurements were taken during the last 36 seconds of second-stage burning, so that aerodynamic disturbances were negligible. The measured roll rates resulted in torques of 0.863, 0.703, and 0.583 ft-lb for the three flights. The calculated possible unbalance of the roll control system in the null position is 0.89 ft-lb. Thus motor induced roll appears to be masked by the roll control unbalance and in the worst case could not be more than the sum of the measured roll and the unbalance (0.863 + 0.89) or 1.753 ft-lb.

4.1.3 Candidate Subsystems Comparison. The propulsion system was selected on a total mission accomplishment basis which reflects the selection criteria stated in the JPL 1971 VOYAGER Mission Description. In order of precedence, these items are: reliability, performance, cost, growth potential to 1975/77, and additional 1971 Mission capability. The tradeoffs relating to each of the candidate solid orbit injection motor configurations for these areas are discussed below.

4.1.3.1 Reliability. The reliability of the orbit insertion motor is closely identified with that of the separate MC/OA system which is required to perform the trajectory correction maneuvers. The aggregate reliability of the two systems comprises the total value for the mission propulsion functions. The reliability of the midcourse system is discussed in Section 4.2, hence, this section involves a direct comparison of the three competing orbit injection motor designs.

4.1.3.1.1 Aerojet Modified Minuteman. The Wing VI Minuteman second stage motor on which the proposed VOYAGER motor is based has successfully passed development, qualification, and acceptance testing and is currently operational with the USAF. The reliability history of this motor is discussed in NOTE 5 of the Classified Supplement.

Most of the testing experience gained on the Wing VI program may be applied to the VOYAGER development program, both on the system and component levels. In terms of the integrated system, the number of static and flight tests conducted since the first Wing VI motor was fired in 1963 may be added to those planned for the VOYAGER development effort. This increases the sample size and thus the confidence level of the observed reliability figures. Also, many of the potential failures due to the interaction of components have by now been eliminated and the VOYAGER development begins with a partially qualified design.

On the component level, in addition to the Wing VI Minuteman development experience, test history is available from the several programs on which technology of the VOYAGER design is based, using either modifications of the designs or the actual components. Here again, this large volume of tests may be added to the sample size to increase the confidence level of the observed reliability figures to be derived from VOYAGER development testing. Also, those failures, which are expected to occur in the early phase of component development, have already done so and have been satisfactorily resolved on the Wing VI program.

4.1.3.1.2 Aerojet Ovaloid. The ovaloid design consists of components, configurations, and materials that have been well characterized in recent development and production programs; principally the Polaris A-3 first stage and also the Wing VI Minuteman. Here, again, this test history could be added to the projected VOYAGER development effort to increase the sample size and hence the confidence level of the observed reliability figures. However, the components in the ovaloid design have not been fired in a similar configuration as is true in the case of the modified Minuteman.

4.1.3.1.3 Thiokol TU-533 and TU-535. The reliability of the Thiokol design is based on development of similar components in the same manner as the Aerojet ovaloid. The principal technological bases are the Surveyor retromotor and the first stage Minuteman, from which the nozzle and safe and arm designs and the propellant formulation are derived. However, the Thiokol designs have neither the production experience of the Wing VI motor nor the background comparable to the Polaris A-3 which is applicable to the ovaloid. Any increase of the sample size to enhance the confidence level of observed reliability derived from VOYAGER testing would be dependent entirely on testing on the component level.

4.1.3.1.4 Summary. Based on the above, the modified Minuteman is considered the most suitable of the three designs, on the basis of having the greatest amount of directly applicable test experience, including firings of motors of similar configuration. This background may be applied to the test experience to be gained during the VOYAGER development effort, thereby increasing the confidence level in the observed reliability.

4.1.3.2 Performance. The candidate motor systems were studied to determine their capabilities to perform the projected missions. The scope of the study, objectives and constraints, analytical techniques and rationale, results, and conclusions are presented below.

4.1.3.2.1 Scope of Study. Because of the interactions of MC/OA and orbit injection propulsion system performance and operations, it was necessary to size both propulsion systems simultaneously using an iterative computer routine. The requirements for MC/OA propulsion using both bipropellant and monopropellant engines were determined for each of the three candidate solid motor designs. The object was to define the bounds of the system in terms of propulsion weight for 1971/73 and available velocity for 1975/77. The 1971/73 Missions are described below and the 1975/77 are described in Section 4.1.3.4.

4.1.3.2.2 Objectives and Constraints. All systems, of course, were required to conform to the following constraints, in addition to the design and performance requirements as given elsewhere in this report.

a. Payload - Combined Spacecraft and lander weight of 5500 pounds, which does not include propulsion.

b. Velocity - Orbit injection 2.0 km/sec minimum with a design goal of 2.2 km/sec.

c. Midcourse - 200 meters/sec (total of all trajectory corrections prior to orbit injection).

d. Orbit Adjust - 100 meters/sec (total of all trajectory corrections after orbit injection).

e. Propulsion Weight - 15,000 pounds total maximum allowable, to be minimized consistent with conservative design practices.

f. Performance - Effective specific impulse and mass fraction values were obtained from preliminary design data and updated as the systems became defined. These data, which were based on currently qualified propellants and design concepts are shown in Tables 4.1-4 and 4.1-5 of the Classified Supplement and Section 4.1.1.1.4 of this document.

4.1.3.2.3 Techniques and Rationale. An iterative computer routine was used to size the propulsion systems taking into account the interaction between MC/OA and orbit insertion operations. Motor designs were optimized by the subcontractors to achieve the minimum weight motors for the velocity and payload data supplied. A fundamental constraint is the fraction of midcourse maneuver assumed prior to orbit insertion as the basis for sizing the solid motor. Early in the study it was decided to assume a full midcourse maneuver (200 meter/sec) and size the solid motor to deliver the required velocity if this maneuver were actually made. The rationale for this is as follows:

a. The total system is some 18% lighter than if zero use were assumed.

b. Execution of less than a 200-meter/sec midcourse maneuver results in an undershoot condition at orbit insertion, after which, the velocity deficiency is compensated for with the orbit adjust system. Use of the undershoot concept is somewhat more conservative from a planetary quarantine standpoint. The variation of orbit injection velocity with the velocity increment of the midcourse maneuver is shown in Figure 4.1-22.

c. Because of the common MC/OA tankage, those propellants which are not used at midcourse are available for orbit adjust maneuvers, increasing mission flexibility. Sizing for the opposite extreme (overshoot) would result in a condition where the amount of overshoot would exceed the nominal orbit adjust capability (100 meter/sec).

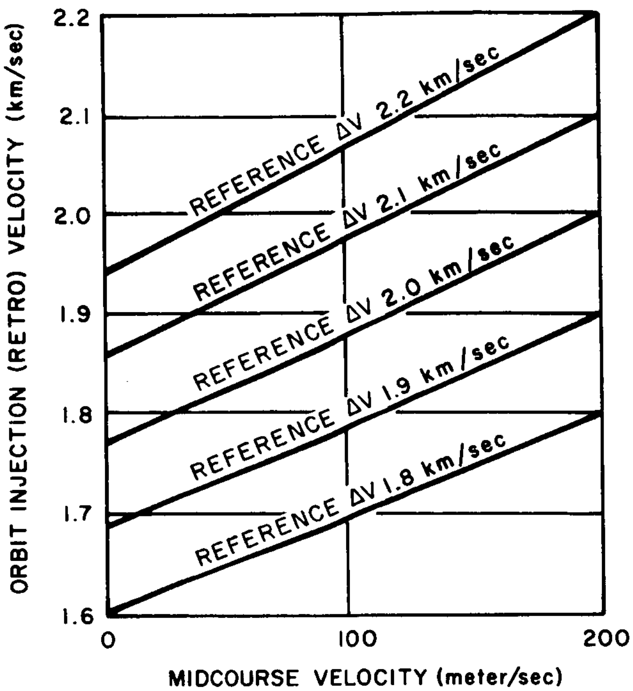


Figure 4.1-22. Effect of Midcourse Maneuver on Orbit Injection Velocity, Aerojet Modified Minuteman 1971/73

d. Trajectory requirements as presently stated call for 1.9 km/sec orbit injection velocity. A system sized for 2.2 km/sec at full midcourse usage will provide 1.94 km/sec with zero midcourse propellant usage.

e. Velocity overshoot at orbit injection results in an exhaust particle orbit decay condition when the orbit adjust maneuver is used to raise apoapsis, which creates a potential planet contamination condition.

4.1.3.2.4 Results

A. Weights and Velocities - The weight comparisons for optimally designed motors for use with the selected monopropellant MC/OA system, for the 2.2 km/sec orbit injection velocity requirements, are given in Table 4.1-6 .

For injection velocities less than 2.2 km/sec, these weight values will vary as shown in Figures 4.1-23 through 4.1-25. An intrinsic penalty is incurred through the use of the monopropellant MC/OA system rather than the bipropellant, because of the lower I_{sp} of the former. This is reflected in increased propulsion weights throughout, as indicated in Table 4.1-7 for the selected Aerojet Modified Minuteman Motor.

TABLE 4.1-6. COMPARISON OF MOTORS FOR MONOPROPELLANT MC/OA SYSTEM 1971/73

Motor	Orbit Injection Weight (lb)	MC/OA System Weight (lb)	Total Propulsion Weight (lb)	Motor Length 1971/73 (in.)	Motor Diameter 1971/73 (in.)	Induced Acceleration 1971/73 (g)	Burning Time 1971/73 (sec)
Aerojet Mod M/M	9513	2181	11694	134.8	52.0	2.73	90
Aerojet Ovaloid	8771	2083	10854	81.6	76.9	2.87	91
Thiokol TU-533	9136	2118	11254	111.7	52.0	3.0	74

B. Envelope - The motor lengths and diameters for the candidate motor designs, assuming a monopropellant MC/OA system are given in Table 4.1-6.

C. Acceleration and Burning Time - The induced acceleration levels for the candidate motor designs are given in Table 4.1-6, again assuming a monopropellant MC/OA system throughout and optimum motor designs.

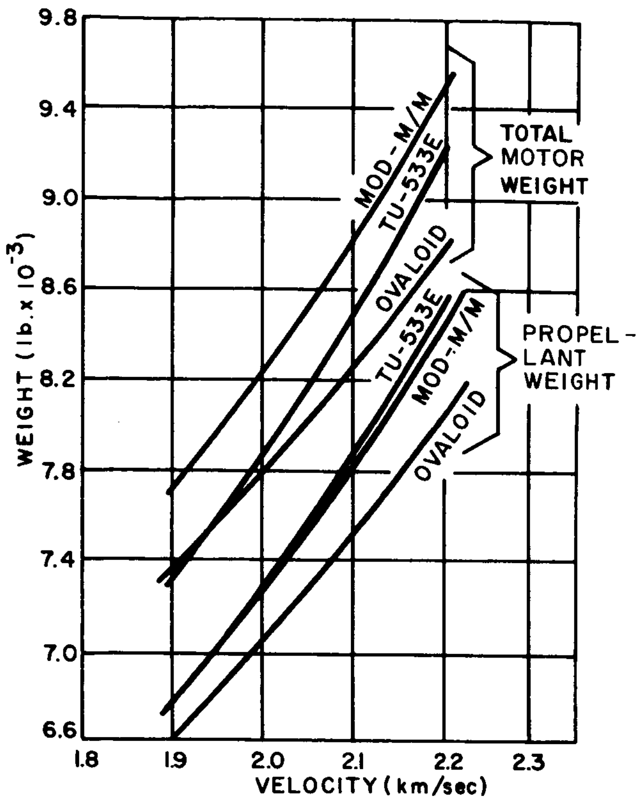


Figure 4.1-23. Orbit Injection Motor Weights Versus Velocity Increments

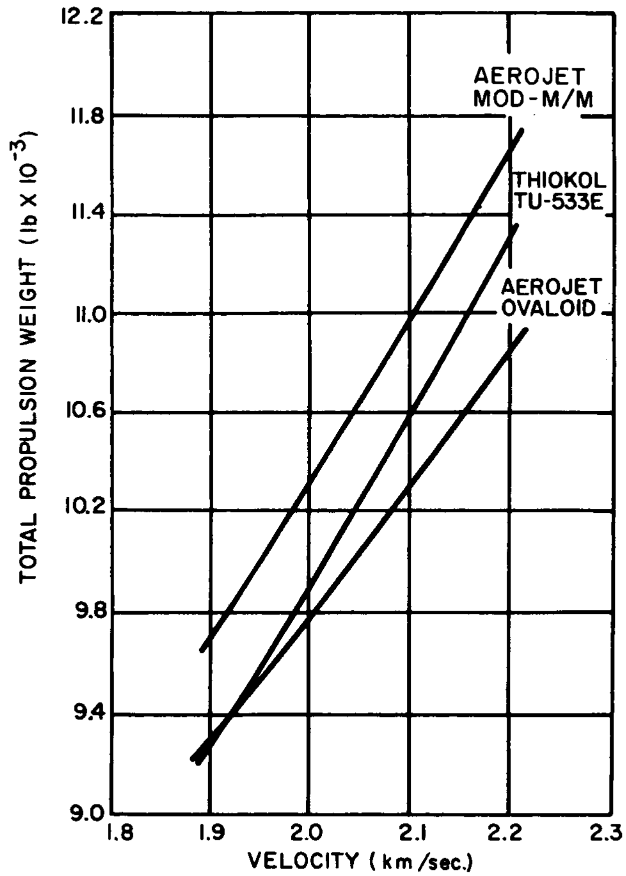


Figure 4.1-25. Total Propulsion System Weight Versus Velocity Increments

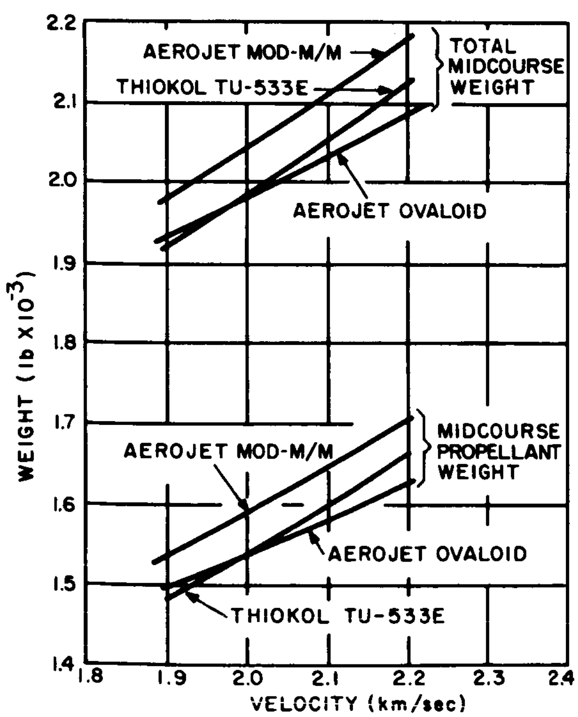


Figure 4.1-24. Midcourse Propulsion System Weights Versus Velocity Increments

TABLE 4.1-7. WEIGHT COMPARISON BETWEEN BIPROPELLANT AND MONOPROPELLANT MC/OA SYSTEMS

MC/OA System	Orbit Injection (lb)	Midcourse System (lb)	Total Propulsion (lb)
Bipropellant	9310	1699	11009
Monopropellant	9513	2181	11694

4.1.3.2.5 Summary. The above tradeoffs indicate that the performance available from any of the candidate systems is sufficient for the requirements of the VOYAGER Mission. Selection of the modified Minuteman with a monopropellant MC/OA fulfills these requirements without significant penalties in propulsion weight in 1971/73. The greater length of the modified Minuteman reduces potential exhaust plume heating problems which may occur with the abruptly terminated nozzle of the Thiokol motors, and potential packaging problems of the large diameter ovaloid are avoided.

4.1.3.3 Cost. Factors which influence the development cost of solid motors were considered in the evaluation, and are discussed below. The total cost of the propulsion function was reviewed on the basis of budgetary cost estimates supplied by the subcontractors for the solid motor and liquid MC/OA systems.

4.1.3.3.1 Aerojet Modified Minuteman. The redesign of existing qualified components, rather than complete new design, will limit component development, propellant reformulation, and static firing efforts. Redesign of the motor to satisfy the 1975/77 Mission requirements will require redesign and tooling modifications and a propellant development and qualification program if beryllium based propellant is used.

4.1.3.3.2 Aerojet Ovaloid. Because of the effort and tooling required to develop a completely new chamber, the need for new casting tooling, and the increased amount of testing required to demonstrate reliability at confidence levels comparable to those of the modified Minuteman, costs would be considerably higher, although some common components would be used. Adaptation of the ovaloid to the 1975/77 Mission would involve the development of a new chamber of larger diameter, plus possibly a beryllium based propellant.

4.1.3.3.3 Thiokol TU-533 and TU-535. The Thiokol motors use Minuteman propellant and some Surveyor components, such as the safe arm and nozzle. However, Thiokol experience on fiberglass chambers is limited to R&D programs of low numbers, and an extensive development and type approval test program would be required. Adaptation of the motor to 1975/77 requirements would necessitate redesign and propellant development along the lines of the modified Minuteman.

4.1.3.3.4 Summary. Based on the configuration similarities between the modified Minuteman and the operational Wing VI motor, the availability of tooling, and the availability of qualified hardware, this motor offers the potential for the lowest total development and delivery costs for the VOYAGER application.

4.1.3.4 Growth Potential to 1975/77. All three designs are adaptable to the 1975/77 Mission requirements. These systems were analyzed subject to the following constraints:

a. Payload - Combined Spacecraft and lander weight of 13500 lb exclusive of propulsion.

b. Velocity - Orbit injection to be maximized.

c. MC/OA same as 1971/73.

d. Propulsion Weight - 15,000 pounds total.

e. Performance - Effective specific impulse and mass fraction values based on preliminary design data and updated as the systems became defined. Advances in performance through the use of beryllium additives in the solid propellant were considered. These data are presented in Table 4.1-2 and NOTE 6 of the Classified Supplement.

4.1.3.4.1 Results

A. Weights and Velocities - The weights of the optimally designed systems, showing the relative apportionment of the 15,000 pound total propulsion weight between the orbit injection and MC/OA systems, are given in Table 4.1-8. Also listed are the attainable velocity increments for each orbit injection motor, using propellant with beryllium additives.

TABLE 4.1-8. COMPARISON OF MOTORS FOR MONOPROPELLANT MC/OA SYSTEM 1975/77

Motor	Orbit Injection (lb)	MC/OA System (lb)	Total Propulsion (lb)	Velocity Increment (kps)	Length (in.)	Diameter (in.)	Acceleration (g)	Burn Time (sec)
Aerojet Mod M/M	11310	3690	15,000	1.485	161.5	52.0	1.82	90
Aerojet Ovaloid	11322	3678	15,000	1.538	96.0	86.0	1.94	95
Thiokol TU-535	11325	3675	15,000	1.493	139.0	52.0	2.41	71

The advantages obtained by using beryllium rather than aluminum additives with either MC/OA system are shown in Table 4.1-9.

B. Envelope - The motor lengths and diameters for the 1975/77 Missions, assuming a monopropellant MC/OA system are given in Table 4.1-8.

TABLE 4.1-9. BERYLLIUM VERSUS ALUMINUM ADDITIVES

MC/OA System		Velocity Increment	
Type	Weight (lb)	Using Beryllium (kps)	Using Aluminum (kps)
Bipropellant	2939	1.588	1.519
Monopropellant	3690	1.485	1.418

C. Acceleration and Burning Time - The maximum induced acceleration and corresponding burning times for the three motor designs, assuming a monopropellant MC/OA system are given in Table 4.1-8.

4.1.3.4.2 Off-loading. The possibility of using the 1975/77 hardware in 1971/73 with a modified grain design to reduce propellant weight was investigated. A performance penalty is incurred, due to the reduced volumetric loading and thus, mass fraction. This is shown in Table 4.1-10 for two configurations:

a. Using hardware designed for use with beryllium based propellant in 1975/77 and substituting a reduced amount of aluminum based propellant, in 1971/73.

b. Using hardware designed for use with aluminum based propellant in 1975/77 and casting a lesser amount of the same propellant in 1971/73.

TABLE 4.1-10. CONFIGURATION COMPARISON

Configuration	Orbit Injection (lb)	MC/OA System (lb)	Total Propulsion (lb)
Be to Al	9893	2230	12123
Reduced Al	9664	2200	11864
Optimum Design	9513	2181	11694

The above data are for the selected Aerojet Modified Minuteman with a monopropellant MC/OA system. The reason for the difference between the "Be to Al" and "Reduced Al" weights stems from the lower density of the beryllium based propellant, which increases inert weights. This increase is reflected in the off-loaded hardware in 1971/73.

The induced accelerations and burning times for 1975/77 motors off-loaded to 1971/73 propellant weights is shown in Table 4.1-11.

TABLE 4.1-11. MOTOR ACCELERATION AND BURNING TIME COMPARISON

Configuration	Induced Acceleration		Burning Time	
	1971/73 (g)	1975/77 (g)	1971/73 (sec)	1975/77 (sec)
Be to Al	3.3	1.82	82	90
Reduced Al	3.3	2.00	80	90

The acceleration may be reduced below 3.0g in 1971/73 by increasing the duration to 90 seconds for the reduced Al configuration, however, use of the same propellant in 1975/77 will extend the duration above 100 seconds, which may be marginal for component integrity.

Further tradeoffs between allowable acceleration, increased motor structural weight to withstand longer burn times and the advisability of changing propellant from 1971/73 to 1975/77 will have to be studied.

4.1.3.4.3 Methods of Increasing Capability. Additional velocity capability may be gained in 1975/77 through use of techniques which are not presently regarded as state-of-the-art, but which are expected to be developed in time for application to these missions. These techniques and their impact on the capabilities of the motors are as follows:

a. Specific Impulse Increase - Specific impulse may be increased by the substitution of beryllium additives for the present aluminum, and this concept is equally applicable to all three motors. Both contractors have, at this time, sufficient altitude performance data from test firings and are expected to have accumulated more by the time development is initiated. If beryllium additives are used, the ovaloid motor is capable of delivering 53 meters/sec more velocity than the modified Minuteman and 45 meters/sec more than the TU-535. Performance may be increased through extension of the exit cone to a higher expansion ratio. However, the increase in delivered specific impulse must be weighed against the added exit cone weight and increased vehicle structure weight due to the increase in length. The latter not withstanding, total orbit injection weight was shown to decrease with an increase in expansion ratio out to 120:1 for the Aerojet motors, however, practical considerations of length limitations resulted in the choice of 70:1 as the design value. Optimization studies of the Thiokol motors showed that a minimum motor weight was achieved at an expansion ratio of about 39:1. Therefore, growth potential for the Thiokol motors would seem limited in this category.

b. Inert Weight Reduction - One possible weight reduction technique would be the substitution of a colombium exit cone for the present ablative unit. Aerojet has successfully fired three motors using colombium exit cones and the Minuteman Wing VI propellant, which is also proposed for the VOYAGER motor. Further development of this component is to be funded under the Minuteman Product Improvement Plan, and a considerable amount of data should be available by the time development is initiated for the 1975/77 motors. Higher thrust vector control performance may be obtained by using nitrogen tetroxide as the injectant rather than Freon. Because of the experience accrued by both subcontractors with Freon, this fluid was selected by both. Either company would be capable of adapting their systems to nitrogen tetroxide for 1975/77, thereby increasing mission capability.

4.1.3.4.4 Envelope Restraints. All motors fit the nominally defined envelope, but packaging considerations with regard to the MC/OA and attitude control system indicate the desirability of a longer motor of smaller diameter. Therefore, the ovaloid, which grows to 86-inches in diameter for the 1975/77 Mission, imposes packaging difficulties which are avoided with either the modified Minuteman or the TU-535, and must be rated inferior to those two in that respect. The TU-535, being some 65.5 inches shorter than the modified Minuteman, yields a decrease in Launch Vehicle shroud weight, but induces problems of plume expansion and solar cell heating due to the abruptly terminated exit cone.

Propellant weight increases for 1975/77 may be accomplished with the modified Minuteman or TU-535, using either beryllium or aluminum propellant, by extending the barrel section of the chamber. In the case of the Aerojet titanium chamber, this requires modification of welding and assembly tooling, and for the Thiokol fiberglass chamber requires modification of the plaster mandrel design and winding techniques. The decrease in motor length obtained by using beryllium rather than aluminum additives is only 5.8 inches for the modified Minuteman, and would be comparable for the TU-535.

4.1.3.4.5 Summary. The modified Minuteman motor appears the most desirable from the standpoint of envelope, is adaptable to beryllium based propellant, can be enlarged without redesign of major components save the barrel section of the chamber, and is capable of delivering a velocity increment only slightly below the other two motors. It rates high, therefore, in the growth potential category in comparison to the other two motors.

4.1.3.5 Additional 1971/73 Mission Capability. The total propulsion weight for the 1971/73 missions is well within the 15,000 pounds allowable for all three candidate motors, as shown in Table 4.1-12.

Extension of the propulsion system to the full 15,000 pounds would result either in an increased velocity increment or additional payload capability in proportion to the values given above.

TABLE 4.1-12. MOTOR CONFIGURATION COMPARISON

Motor	Total Propulsion (lb)	Growth Potential (lb)
Aerojet M/M	11694	3306
Aerojet Ovaloid	10854	4146
Thiokol TU-533	11254	3746

The techniques described for growth potential could also be applied to increasing 1971/73 Mission capability, but to do this would increase schedule risk and cost and possibly imperil the objective of mission accomplishment. The extent to which this would apply would vary, as indicated by the growth potential discussion for each component and motor.

4.1.3.6 Conclusion. The modified Minuteman was rated superior to either of the other two designs on the basis of the selection criteria given above. The principal considerations affecting this rating are summarized herein.

The clear-cut advantage of this motor in the primary selection criterion of reliability stems from the similarity of the configuration to the currently operational Wing VI Minuteman second stage motor and the extensive component historical base on which the VOYAGER motor development effort may be founded. This will increase the confidence level of the observed reliability figures obtained from VOYAGER testing.

The performance capabilities of the modified Minuteman, in terms of total propulsion weight in 1971/73 and velocity increment for 1975/77, are slightly lower than the other two motors. However, an ample weight margin exists in the first case and in the latter the attainable velocity is within the range of practical trajectories for these missions.

4.2 Midcourse/Orbit Adjust (MC/OA) Subsystem

4.2.1 Introduction. Selection of a MC/OA propulsion subsystem for use in conjunction with a solid retropropulsion unit must be made on the basis of reliability, performance, cost, development risk, and growth potential. Based on these criteria, propulsion concepts such as restartable solids, throttlable solids, "cap pistol" systems, multiple solid rockets, cryogenic liquids, and advanced high performance liquids were immediately eliminated. The candidate systems left for further consideration were: (a) a hydrazine blend fuel and nitrogen tetroxide oxidizer bipropellant system and (b) a monopropellant hydrazine system. To assure that the latest technology in propulsion systems was used in making this selection, designs for a MC/OA system were solicited from major propulsion system suppliers. In the bipropellant area, studies were requested from Aerojet General/Liquid Rocket Operation using their 2200 lb engine, Bell Aerospace based on their 100 lbf thrust chamber, Rocketdyne based on their work in space engines for Transtage, Gemini and other programs, the Marquardt Corp. for the 100 lbf thrust chamber used in the RCS on the Apollo LEM vehicle, and Thiokol Chemical, Reaction Motor Division for their 100 lbf C-1 thrust chamber. In the monopropellant area, Rocket Research Crop. and TRW Propulsion Group were asked to conduct studies.

Based on these studies and substantial in-house effort, designs for each type of system were evolved and are represented schematically in Figures 4.2-1 and 4.2-2. Both of these approaches were studied in detail in the light of the original criteria, and were expanded to include number of thrust chambers, burn time, thrust vector control, propellant acquisition under zero-g conditions, slosh control of the propellant in the tank, and long life in a space environment.

4.2.2 Subsystem Description. A detailed description of the monopropellant system is contained in VC238FD101, Volume A. In summary, it is a pressurized system using regulated helium gas as the pressurant. The propellant, hydrazine, is stored in four tanks which use butyl rubber bladders for positive propellant expulsion. Propellant control of each of the four thrust chambers is provided by quadredundant solenoid valves. When the hydrazine is forced into the thrust chamber, it is decomposed by the Shell 405 spontaneous catalyst into gases at approximately 1800°F. Banks of squib actuated valves isolate pressurant and propellant during coast periods and for final lockup.

The design of the bipropellant system is based on the same philosophy that was used for the monopropellant system, hence the schematics are similar. The differences are: (a) quadredundant check valves are added in the pressurant lines feeding each propellant, to minimize possible mixing of propellant vapors in the lines; (b) normally open explosive valves are added in the pressurant lines to each tank to provide positive system lockup upon completion of the mission; (c) identical explosive valve banks are used in each propellant line; (d) a two plane gimbal and actuator system is provided on each thrust chamber for pitch, yaw, and roll control; (e) and supplemental telemetry, fill valves, and pressurant valves have been added as required. On the basis of these two systems, detailed tradeoffs have been made in arriving at the selection of the preferred system.

4.2.2.1 ΔV Capabilities, Accuracy, and Thrust Level Selection. Desired minimum ΔV for midcourse maneuvers is 0.1 meters/second. Thrust level for the MC/OA propulsion system

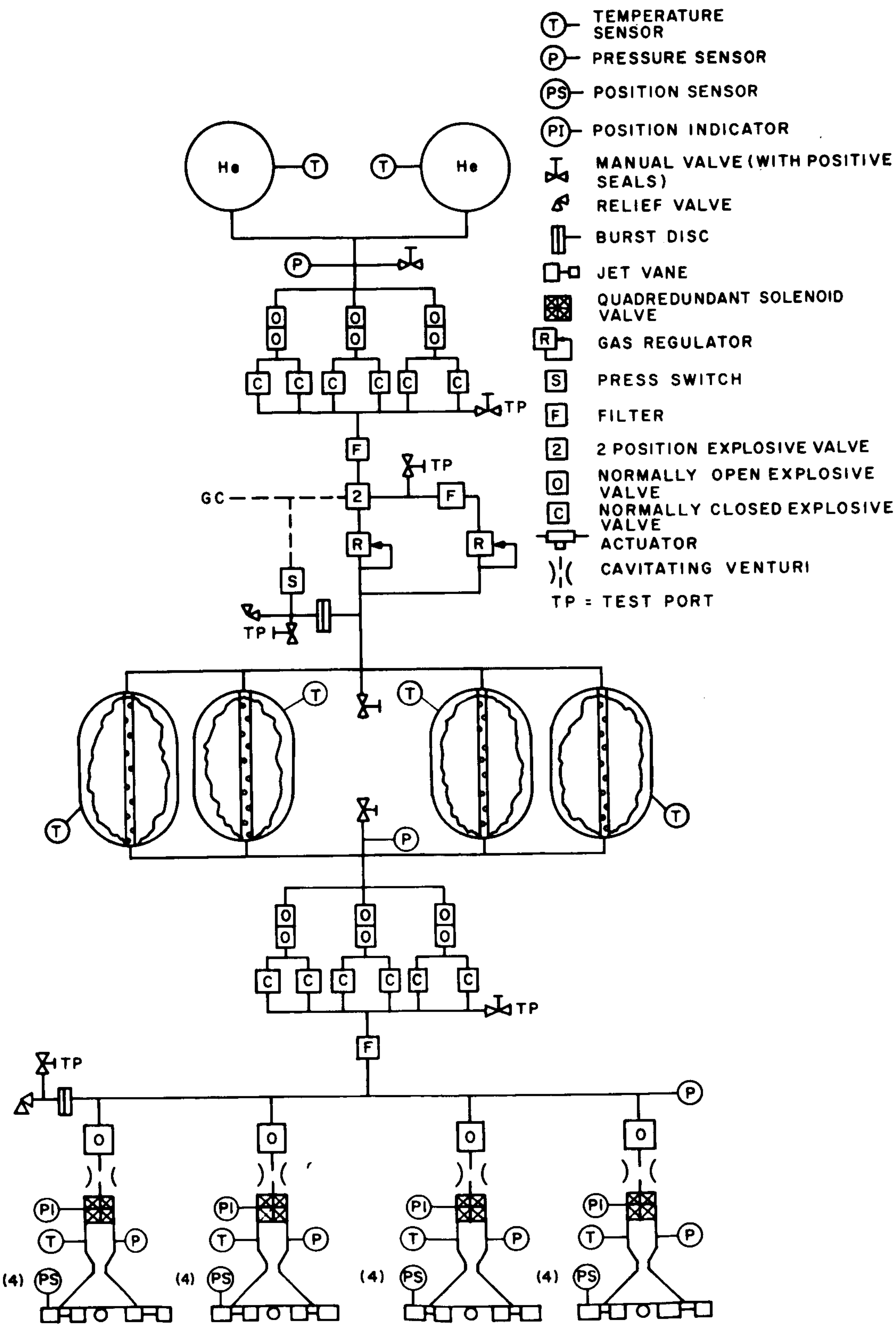


Figure 4.2-1. MC/OA Propulsion System, Monopropellant, Schematic

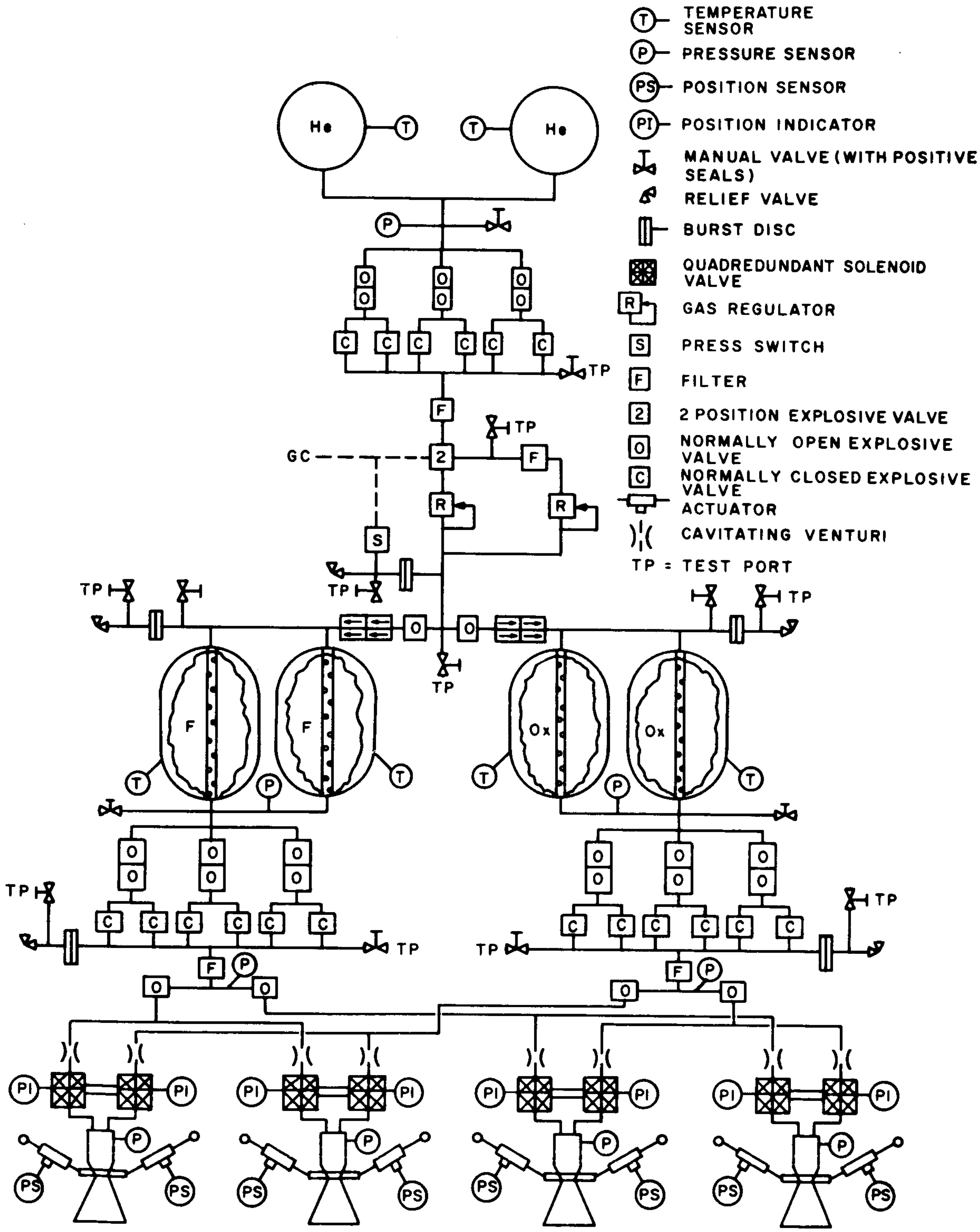


Figure 4.2-2. MC/OA Propulsion System, Bipropellant, Schematic

must trade off minimum ΔV capability, burntime, and state-of-the-art in thrust chamber technology. It is obvious that a low thrust level of 50 pounds could achieve minimum ΔV's far less than the desired values. A number of developed thrust chambers, both monopropellant and bipropellant, are available in this range. However, total burntime for a 100 meters/second midcourse correction would approach 3500 seconds or nearly one hour. This is not only a long burn time from the propulsion standpoint, but it is also a source of appreciable guidance inaccuracies due to time dependent errors.

Based upon experience with existing hardware, a thrust level of 100 pounds can be considered about the upper thrust limit for either monopropellant hydrazine or radiatively cooled bipropellant thrust chambers. To achieve thrust levels more compatible with guidance requirements, multiple thrust chambers must be considered. Although odd numbers of chambers could be used, it is more practical to use pairs oriented to give symmetry about one or more of the major control axes. Guidance analyses have shown that either 200 or 400 pounds thrust offers reasonable compromise between short burn time and long burn time inaccuracies. Data obtained from Rocketdyne indicates the minimum ΔV's obtainable with four 100-pound thrust chambers will vary with spacecraft weight as shown in Figure 4.2-3. If the minimum ΔV of 0.1 meter/sec is to be met, propulsion capabilities will limit maximum thrust to about 400 pounds.

The desired propulsion system accuracy in making the minimum ΔV maneuver is ±0.007 meters/second. Data from Rocketdyne for the four 100-pound thrust engines is shown in Figure 4.2-4. Propulsion accuracy is seen to be well below this limit, even for orbit adjust maneuvers.

4.2.2.2 Packaging Envelope. Use of a single solid propellant motor for retropropulsion effectively blocks the installation of MC/OA thrust chambers along the primary roll axis. A single MC/OA thrust chamber oriented to another axis presents two major problems. First, the shift in cg of the Flight Spacecraft before and after retropropulsion may dictate a major reorientation of the thrust axis for orbit adjust maneuvers. This can be done by employing extremely large gimballing capabilities or, perhaps, by having two operating positions for the thrust chamber. The second problem involves guidance and control. Multiple thrust axes require added complication for the autopilot. It greatly simplifies control requirements if all maneuvers (midcourse, retro, and orbit

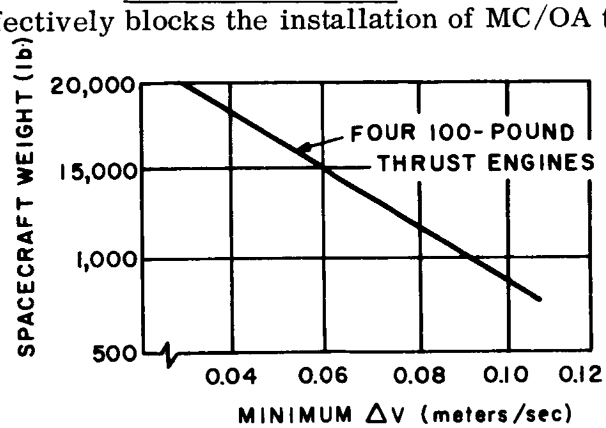


Figure 4.2-3. Minimum ΔV Obtainable with Four 100-Pound Thrusters

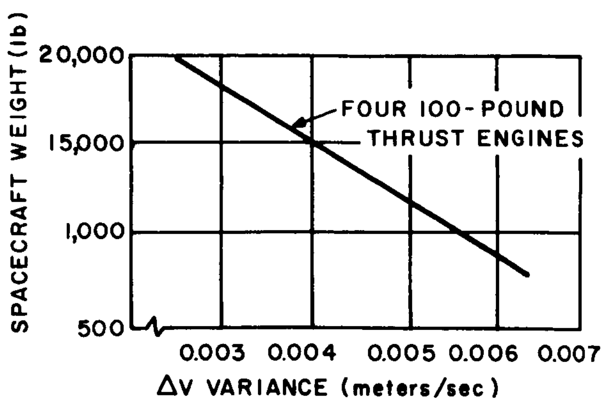


Figure 4.2-4. ΔV Variance with Four 100-Pound Thrusters

adjust) can be made along a single axis. To eliminate these difficulties, multiple MC/OA thrust chambers should be used.

Based on the previously discussed thrust levels and chamber sizes, four 100-pound thrust chambers, equally spaced about the roll axis and oriented to the pitch and yaw axes, appear to satisfy all propulsion, guidance, and reliability requirements. This configuration has an added advantage that, in the event of a thrust chamber malfunction, an opposing pair of chambers may be shut down and the remaining pair used to carry out any further maneuvering requirements.

The complete MC/OA propulsion system must be packaged within the volume bounded by the 52-inch diameter solid case and the chosen 120-inch diameter propulsion system restriction. No problem is apparent in packaging either a monopropellant system or a bipropellant system within this annulus. With a monopropellant system, a choice must be made between a large number of spherical propellant tanks (six or eight) and a fewer number of cylindrical tanks. Since use of bladders with hydrazine is not substantially more difficult with cylinders, the choice would favor a fewer number of cylindrical tanks to reduce the amount of plumbing required. Four tanks with four thrust chambers makes a logical packaging configuration.

4.2.2.3 Propellant Control

4.2.2.3.1 Zero-g Acquisition. Either type of MC/OA system must incorporate a means for positive propellant acquisition under zero-g conditions. The use of butyl rubber bladders in monopropellant hydrazine systems is a proven technique for long space missions. A problem area exists if the propellant and bladder temperature exceeds 100°F, since the rubber will then begin to react, blistering and discoloring the propellant. However, this does not preclude its use on VOYAGER, since propellant temperatures should not reach this point under normally expected conditions.

In a bipropellant system, the blended hydrazine fuel may be contained in butyl rubber bladders as in the monopropellant system. The major problem is the oxidizer. No completely satisfactory bladder material has been developed for nitrogen tetroxide. Aluminized teflon has been used with some success in a few applications. These aluminized teflon bladders can by cycled a number of times without damage but they do permit some permeation of oxidizer through the material. A number of other methods were suggested by various propulsion suppliers. These included convoluted metallic diaphragms, rolling metal diaphragms, and zero-g screens. Based on the experience available and the anticipated propulsion system acceptance and testing cycles, aluminized teflon bladders would be tentatively selected although further development work is needed.

4.2.2.3.2 Sloshing. If a standpipe is used within the bladders, as is normally the case, and if the standpipe is anchored or restrained top and bottom, transverse movement of the liquid will be restricted. When the propellant is contained wholly within the bladder, damping of fluid motion comes primarily from the material characteristics of the bladder. If maximum damping is required, fluid should be stored between the bladder and the tank, i.e., when the tank is fully loaded with propellant, the bladder is collapsed about the standpipe. As propellant is used, the bladder expands forming a bubble within the main fluid mass. Motion of this bubble is restrained both by the bladder characteristics and the resistance of the fluid. Thus practically no sloshing will occur. Either of these methods apply to monopropellants and bipropellants.

4.2.2.4 Maneuvering Capabilities

4.2.2.4.1 Thrust Vector Control. Monopropellant systems normally achieve thrust vectoring by means of jet vanes although gimballing could be used if desired. Jet vanes and actuator assemblies developed on the Mariner Program have proven their reliability under space conditions almost identical to those required for VOYAGER. Thus, for a monopropellant system, the best choice for thrust vectoring is the existing jet vane and torque motor actuator systems.

Because of their higher operating temperatures, bipropellant systems do not normally make use of jet vanes. Instead, full gimballing is required. Gimbal systems actuated by torque motors should be satisfactory at these low thrust levels. However experience with gimbaled 100-pound thrust chambers is limited, particularly in the area of flight proven systems.

4.2.2.4.2 Roll Control. One of the advantages of the jet vane or the fully gimbaled systems is that roll control can be achieved with either of the systems without modification. Present information indicates that roll control (beyond that available from the ACS system) is not needed with the solid propellant system. However if such a need should arise, either type of MC/OA system could provide this capability without difficulty.

4.2.3 Subsystem Comparisons. To arrive at a selection of the preferred system, the two candidates, monopropellant and bipropellant, were compared on the basis of reliability, performance, cost, growth potential for 1975/77 Missions, and added capability in 1971. Because both systems are essentially new designs, and therefore little or no system test data or experience exists, the arguments tend to be more qualitative than quantitative. This is particularly true in the case of reliability where generic failure rates must be used in lieu of test data to calculate a numeric probability of success. Employing generic data is misleading since the method of application deviates widely compared to the VOYAGER application. Generally, this situation exists in all areas in which comparisons were made.

4.2.3.1 Reliability. In the design of both systems, reliability was considered to be the prime design criteria. Therefore, those areas or components that were considered to have a low relative reliability or were critical to mission success were made redundant. Some of these are:

a. Manual Valves - sealed off prior to launch by caps, welding, or brazing to provide redundant sealing against leakage.

b. Sets of normally open explosive valves - used in series to provide redundant sealing at isolation.

c. Sets of normally closed explosive valves - used in parallel to provide redundant opening at activation.

d. Pneumatic Regulator - used in parallel with one on standby.

e. Quadredundant solenoid valves - used to provide series/parallel paths for shutdown/start propellant control.

f. Burst Diaphragm in Series with Relief Valve - to provide redundancy in sealing leakage while providing a safety feature to the system.

g. Four Thrust Chambers - provides for "pair out" capability.

Because of this redundancy in critical areas, a numerical calculation of reliability would indicate a high and almost equal probability of success for either system. However, a review of the schematics of both systems (Figures 4.2-1 and 4.2-2) shows that the bipropellant system contains a great many more components, plumbing lines, and connections to cause problems in flight operations, ground test, or system servicing. Therefore, it was concluded that the monopropellants had an advantage in this area.

In comparing the two systems on the basis of reliability, one of the most important considerations and hardest to evaluate is the method of assuring positive propellant acquisition. In the case of monopropellants, the Ranger and Mariner vehicles demonstrated the feasibility of butyl rubber bladders.

Bladders have also been used with bipropellants, but the mission durations have been on the order of hours or days compared to months for VOYAGER. Generally the materials for bipropellant bladders are FEP and TFE teflon used in successive layers. One problem appears in the oxidizer section, where the nitrogen tetroxide permeates the material such that over an extended period there could be a considerable loss of capability caused by oxidizer being trapped on the upstream side of the bladder. A second problem is that they are more cycle limited than the butyl types. When teflon bladders are collapsed, three corner folds result and if the bladder is repeatedly cycled, the material tends to fold in the same pattern and the highly stressed three-corner folds develop holes or tears. This condition is aggravated by operation at the lower temperature limits. If the tears become large enough, the capability for propellant acquisition is destroyed. Controlling the number of cycles (ground testing) would reduce this mode of failure.

These two problems (permeation and bladder cyclic failure) are essentially eliminated using a monopropellant in butyl bladders. Mission reliability is enhanced and ground testing is not limited. Therefore, the monopropellant system was considered much better, from a reliability standpoint, in this area.

Another area to consider as part of a reliability discussion is the timing of arrival of propellants at the thrust chamber injector (catalyst bed). This also relates back to the number of components and propellant acquisition problems. In bipropellant systems, it is usually required that the fuel and oxidizer be injected into the combustion area in a predetermined sequence. This and an associated close time tolerance are required if severe pressure spikes are to be avoided and smooth stable combustion is to be achieved. To maintain uniform and stable combustion in all thrust chambers, flow of both propellants must be maintained. Loss or interruptions of the flow balance or timing due to gas bubbles in the line (bladder failure) or changes in dynamic characteristics (component failure) can easily lead to catastrophic results in the MC/OA system and, perhaps the Spacecraft.

Monopropellants, on the other hand, are not as severely affected by interruptions in the flow process since decomposition (combustion) is based on contact of the hydrazine with the catalyst rather than the mixing of two liquids. Further, decomposition is accelerated by a hot catalyst bed, thus decomposition is easily established following a flow interruption.

Here again, the monopropellant system with its fewer possible failure modes indicates a higher reliability and is therefore, preferred.

In discussing reliability, technical and schedule risk must also be evaluated, considering technical risk first. With monopropellant systems in the "flight proven" category, the greatest technical risk must, of necessity, fall on the bipropellant system. A further analysis of the risk points to the positive expulsion device in the oxidizer tanks as the chief contributor. However, neither system presents a real technical risk.

In looking at schedule risks, it appears that because of the simpler system and more off-the-shelf components, the monopropellant system could be qualified at an earlier date. However, the bipropellant system could be qualified in time to be compatible with overall system schedules.

Flight Acceptance test specifications require testing at 20°C above and 20°C below the 95th percentile of the operating temperature limits. This is interpreted to be 4 to 116°F. Neither monopropellant nor bipropellant systems can meet either of these limits. Monopropellant hydrazine has a freezing point of approximately 35°F and becomes incompatible with the butyl rubber bladder at about 100°F. Bipropellant systems have similar limitations. Bladders in the fuel side will probably begin to react at about the same temperature if hydrazine forms any part of the fuel blend. At the lower end of the temperature scale, nitrogen tetroxide freezes at 11.8°F and Aerozine -50 at 18°F. Thus bipropellant systems show a slight advantage at the low temperature end. However, neither system has an advantage at elevated temperatures.

Neither monopropellant hydrazine systems nor bipropellant hydrazine/nitrogen tetroxide systems have demonstrated a life expectancy under space conditions which approaches the thirteen months required on the VOYAGER Missions. Hydrazine systems built for Mariner were designed to meet requirements approaching this period and there is every indication that the long duration space requirements can be met. The oxidizer is probably the more critical of the two propellants. Although space storage data is lacking ground, storage of nitrogen tetroxide in flight tanks has been demonstrated for periods up to two years. There is no reason to believe that either of the systems cannot meet long term space storage requirements but adequate demonstration will be required.

4.2.3.2 Performance. Both monopropellant and bipropellant systems meet all the performance requirements of:

a. 200 meters/sec midcourse ΔV
b. Minimum impulse bit and accuracy of 0.1 ± 0.007 meters/sec.
c. 100 meters/sec orbit adjust ΔV
d. Propulsion system weight (MC/OA plus retro) is under 15,000 lb.

Weights of loaded monopropellant and bipropellant systems will vary inversely as their respective specific impulses for high total impulse systems. Monopropellant hydrazine has a vacuum specific impulse of about 240 seconds at an expansion ratio of 100. Bipropellant hydrazine/nitrogen tetroxide systems can, at the present time, provide a vacuum specific impulse of about 300 seconds. Thus, monopropellant systems will always weigh about 25% more than an equivalent bipropellant system. Loaded weight of a monopropellant system to meet 1971/1973 mission requirements is 2181 pounds and burnout weight is 475 pounds. However for the 1971/1973 Missions overall propulsion weight (retro plus MC/OA propulsion) totals about 11,691 pounds - well below the 15,000 pound limit.

4.2.3.3 Costs. Estimated costs for bipropellant and monopropellant systems indicate a very small difference in favor of the monopropellant.

4.2.3.4 Growth to 1975/77. From typical parametric studies, the added weight of a monopropellant system for the 1975/77 Missions would penalize the available retropropulsion velocity by about 100 meters/sec (assuming 15,000 pounds maximum propulsion weight).

Of the two systems, the monopropellant system is more limited in growth potential than the bipropellant system. Present technology can give a vacuum specific impulse of about 240 seconds with monopropellant hydrazine. Additives for hydrazine are being tested which may raise this by 5 to 10% but these are still in the research stage. Other monopropellants have already been demonstrated with performance comparable to the nitrogen tetroxide/hydrazine bipropellant systems. However at this time they still have major drawbacks which would preclude their use from a reliability aspect.

A number of bipropellant combinations exist which will give performance 20 to 30% higher than the nitrogen tetroxide/hydrazine system. Most of these however involve at least one cryogenic fluid. Although laboratory work has already demonstrated the feasibility of long term cryogenic liquid storage, it will require better manufacturing and insulating techniques to make long term space storage practical.

4.2.3.5 Added 1971 Capabilities. Since both systems meet the basic design requirements this is not a valid point for comparison.

4.2.4 Tradeoff Summary. A rating system was used to evaluate the two candidate systems (Table 4.2-1). Ratings were:

- -1 = deficient areas
- 0 = average or equal capability areas
- +1 = definite advantage

On this basis the monopropellant system shows a slight advantage, especially when viewed in light of the 1971/73 requirements.

TABLE 4.2.-1. EVALUATION OF MONOPROPELLANT AND BIPROPELLANT SYSTEMS

		Monopropellant System	Bipropellant System
Performance	Weight	0	+1
	ΔV capability and accuracy	0	0
	Packaging envelope	0	0
	Propellant control - acquisition	+1	0
	sloshing	0	0
	Maneuvering capability - TVC	0	0
	roll control	0	0
	Temperature limitations	-1	-1
	Long life and space storage	+1	0
Reliability		0	-1
Growth Potential		0	+1
Development Risk -	Technical	+1	0
	Schedule	0	0
	Cost	0	0
Overall System Rating		+2	0

4.3 Transtage

4.3.1 Existing Transtage

4.3.1.1 Description. The Transtage is an upper-stage Standard Launch Vehicle designed to operate above 350,000 feet in altitude. The propellants are nitrogen tetroxide and Aerozene-50, which are hypergolic, earth storables. The engine includes two identical thrust chamber assemblies. Propellant flow to each assembly is controlled by a fuel pressure activated, pilot operated bipropellant valve. The propellants are stored in titanium tanks which contain baffles for slosh control and propellant traps and screens to aid in propellant acquisition during zero-gravity engine starts. (In the flight tests to date, this method has never been used. Ullage settling was provided by operation of the ACS thrust chambers.) Helium, stored in titanium spheres at ambient temperature, is used for propellant tank pressurization.

Propellant tank pressure is controlled by a set of quad redundant solenoid valves. The valves are activated by two pressure switches that sense pressure in an accumulator common to the fuel and oxidizer pressurant feed lines. (See Figure 4.3-1). Each switch contains series-parallel pressure sensing and switching elements. The pressurant feed lines contain series check valves to prevent propellant mixing and an orifice for balancing propellant tank pressures. Thrust vector control is achieved by two-axis hydraulic actuator gimbaling of both thrust chambers.

4.3.1.2 Performance. The Transtage propulsion module is a multistart system with over 6.5×10^6 lb-sec total impulse capability. The dry and burnout weights are approximately 2290 pounds and 2510 pounds, respectively. The total consumable propellant load is approximately 22,800 pounds. Thrust chamber performance is as follows:

Thrust - 8000 pounds each chamber = 16,000 lb total
Minimum specific impulse = 298 sec
Nominal specific impulse = 305 sec
Minimum impulse bit = 6650 lb-sec
Shutdown impulse uncertainty = ± 780 lb-sec
Start transient asymmetry (roll torque) = 1240 ft-lb-sec*
Shutdown transient asymmetry = 935 ft-lb-sec*

The thrust chambers are qualified to operate with chamber pressures ranging from 90 to 120 psia and with propellant temperatures of +45 to +90°F. The propulsion system was designed for a minimum life in earth orbit of 6.5 hours. Studies have shown that this life could be extended to 30 days with only minor modification to the propulsion module.

4.3.1.3 Development Status. The planned development program for the Transtage includes 17 flights. Most of these will include three starts and one flight is scheduled for ten starts. Six of the flight tests have been completed.

---

*Uncontrolled, actual values would depend on autopilot and gimbal response and would be less than shown.

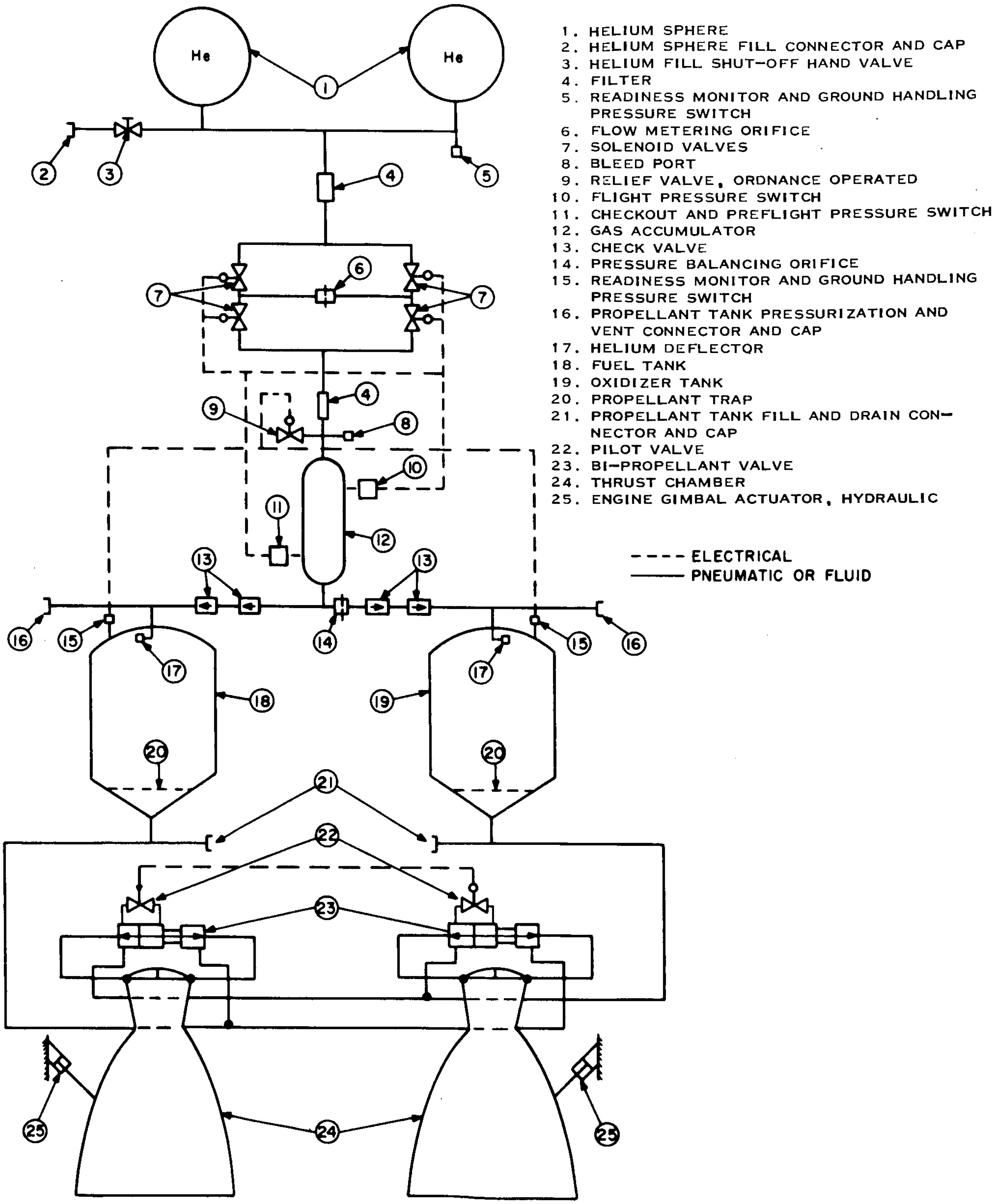


Figure 4.3-1. Transtage Propulsion System (Unmodified)

A flight configuration test unit has been fired in a vacuum chamber for four test runs. The test series included seven engine starts and 830 seconds of total operation over a duty cycle involving a coast period of 7.5 hours.

Other test experience includes ten "battleship" configuration firings using flight types of thrust chambers, pressurization subsystem, and propellant outflow and pressurization systems. With the exception of the thrust chamber, all functional components and/or subsystems have completed flight qualification test programs. The thrust chamber has completed its 14 PFRT firings.

4.3.2 Application to the VOYAGER Mission

4.3.2.1 Existing Vehicle Configuration. The Transtage is designed for 6.5 hours of operation in earth orbit. Studies have indicated that certain modifications will be either required or desirable to increase the space life to 300 days. The following is a brief discussion of the required and proposed revisions:

a. Brazed, or whenever practical, welded plumbing joints - To improve pressurization subsystem reliability.

b. Micrometeoroid protection - The forward portion of the vehicle will be protected by the space bus, but the aft section will require the addition of a micrometeoroid shroud to protect the propellant tanks.

c. Thermal control - Will be supplied by the space bus thermal control system and by passive radiation shielding included with the micrometeoroid shroud.

d. Hydraulic actuator leakage - Hydraulic fluid evaporation causes drying of the seals on sliding surfaces of the piston rods. This will be eliminated by adding pressurized boots to cover the piston rods. Brazed fittings and seal terminations will further reduce leakage potential. The motor-pump is already encapsulated and pressurized but, the fluid reservoir volume will be increased to accommodate anticipated leakage.

e. Thrust chamber bipropellant valve - Two design possibilities are available. The valve can be redesigned with redundant seals and possibly redundant series - parallel type of valving, but this will be a major redesign including engine requalification. Prevalves could be placed in the propellant lines to provide near-zero leakage when the engines are inoperative. The existing bipropellant valves would be used to start and shut down the engine. The prevalves would be positioned by electric motor drives operating with high mechanical advantage to ensure positive sealing.

f. Decreased minimum impulse - The minimum impulse required for small velocity changes is less than 500 pound-seconds. Since the Transtage engines are not capable of this small impulse, vernier engines will have to be used to provide for small velocity changes.

g. Propellant acquisition - The present system utilizes propellant traps with check valves to retain sufficient propellant for engine start. For improved propellant acquisition it is recommended that a system of small mesh screens be used to separate the tanks into zones of propellant, propellant-gas, and gas. Use of the screen system will inhibit slosh in the liquid zone. Baffles will be used to reduce propellant movement in the gas/liquid volume.

h. Pressurization solenoid valve leakage - The existing solenoid valves are flight-qualified at a maximum leakage rate of 1000 scc/hr. This leakage could be tolerated, since the propellant tanks will be about half ullage and will be ground pressurized to store about 11 pounds of helium in excess of the required amount of pressurant. The excess helium would allow for over 6000scc/hr leakage during a 200-day mission. The 3-sigma leak rate from the present pressurization system is about 3000 scc/hr.

4.3.2.2 Alternate Methods

4.3.2.2.1 Configuration Changes. Since the VOYAGER Mission will require only about half of the total impulse available, two broad areas of modification were available. First, the propellant tanks may be off-loaded, and, second, the tanks may be shortened. If the tanks are off-loaded the blow-down method of operation is suggested since the ullage volume can be used to store most of the pressurant gas with the remainder stored in smaller spheres at moderate pressures (800 to 1000 psia). The pressure control system would consist only of squib valves to dump the stored helium into the main propellant tanks during the retro portion of the mission. This would result in a highly reliable pressurization system and eliminate the problems of storing pressurant gas at high pressures.

The second area, shortening the tanks, is further subdivided into two major avenues of approach. First, both tanks may have lengths of the cylindrical section removed so that the resulting internal volume meets the propellant requirements. However, in this approach the oxidizer tank becomes limiting because: (a) with all the cylindrical sections removed (leaving only the elliptical forward dome and the conical aft closure butted together) the tank volume is still larger than required and, (b) removal of more than 15 inches of cylindrical section requires major redesign of the tank support structure. These severe conditions led to the second subdivision, which is to use two fuel tanks (modified by changes in cylindrical section length) to tank both fuel and oxidizer. On the basis of preliminary analysis, a modification of this type appears to be less involved than the complete redesign of the oxidizer tank.

In summary, three basic propellant tank and structure configurations were selected for further evaluation; they are:

a. Existing tankage and tank support structure.

b. Propellant tanks shortened 15 inches, with the existing tank support structure retained.

c. Two fuel tanks sized to hold VOYAGER 1975/77 Mission propellant loads with redesigned tank support structure.

4.3.2.2.2 Functional Considerations. In addition to the basic propellant tank and structure configuration modifications previously described, the following tradeoff areas were considered for each configuration:

a. A single main engine versus the existing two main engines to save weight and increase reliability.

b. An oxidizer to fuel weight mixture ratio of 1.6 versus the use of the existing 2.0 mixture ratio. The 1.6 mixture ratio results in equal volume outflow which could save vehicle weight and reduce the length.

c. A squib-valve-initiated blow-down pressurization system versus the existing regulated pressurization system. This will improve system reliability, since parallel squib valves would be the only pressurization components, whereas the existing system consists of switches, relays, check valves, and solenoid valves.

d. Use of the Transtage for retro velocity change only with a separate MC/OA thrust chamber for all other maneuvers versus Transtage for all retro, midcourse, and orbit adjust velocity changes. A change of this type provides for greater accuracy in obtaining minimum ΔV's.

e. Use of the Transtage for retro and large midcourse maneuvers and a midcourse and orbit adjust system with separate tankage, controls, and thrust chambers for the remaining small midcourse and orbit adjust velocity changes. This latter system was considered, since it could also provide propellant settling before Transtage engine start. Monopropellant hydrazine and bipropellants were both considered for the MC/OA functions.

Combining these tradeoff areas with the three basic propellants tank and structure configurations results in well over 100 systems. Table 4.3-1 summarizes the salient arguments used in discarding various configurations.

4.3.3 Candidate Subsystem Comparison. The resulting five configurations and their identifying nomenclature are as follows:

Nomenclature	Configuration
U-E	Existing Transtage with the addition of propellant acquisition screens, vernier thrust chambers, and propellant feed line prevalves.
U-E3	Same as U-E except with a blow-down pressurization system.
MC-3	Same as U-E3 except the propellant tanks are shortened by approximately 15 inches.
MC-1	Two short Transtage fuel tanks using existing internal tank structure, redesigned propellant tank and engine support structure, one main engine, four vernier engines, and existing pressurization system with squib valve isolation.
MC-2	Same as MC-1 except with two main engines.

TABLE 4.3-1. SALIENT REASONS FOR REDUCING CONFIGURATIONS

Areas Eliminated	Major Reasons for Parameter Elimination
1.6:1 Mixture ratio	a. Equal volume tanks resulted in a further displacement of the center of gravity. b. Weight was not saved since the volume density is reduced. c. Engine requalification would be required. d. Possible engine injector redesign.
Single main engine with existing propellant tank and short version of existing tanks	a. Would require major structural redesign for engine support truss. b. Engine burn time would increase, thereby reducing reliability.
Regulated pressurization system with existing or short version of existing propellant tanks except the "as is" Transtage was retained	a. Reliability can be increased by elimination of the regulated pressurization system. b. Weight can be decreased by storing pressurant in the propellant tanks. c. Space storage compatibility is increased by elimination of the largest single source of pressurant leakage.
Separate MC/OA propulsion systems.	a. Reliability was decreased due to the addition of separate propulsion system. b. Studies indicated that propellant acquisition could be achieved without propellant settling.

As may be noted, configuration MC-1 retains the single thrust chamber which was a basis for elimination of all the other configurations where it was considered. The major reason for elimination of these single thrust chamber configurations with existing transtage propellant tank designs was to prevent redesigning the basic vehicle structure. However, configurations MC-1 and MC-2 were considered for detailed study to determine the feasibility of a minimum weight vehicle consisting of Transtage components. Since redesign of the engine support structure is required in either MC-1 or MC-2, the minimum weight system is obviously MC-1. Thus, MC-2 was dropped from further consideration, leaving four primary systems.

Configuration U-E3 will be discussed as the blow-down pressurization candidate. There is very little difference between U-E3 and MC-3 and a complete system design study would be required to determine the optimum configuration.

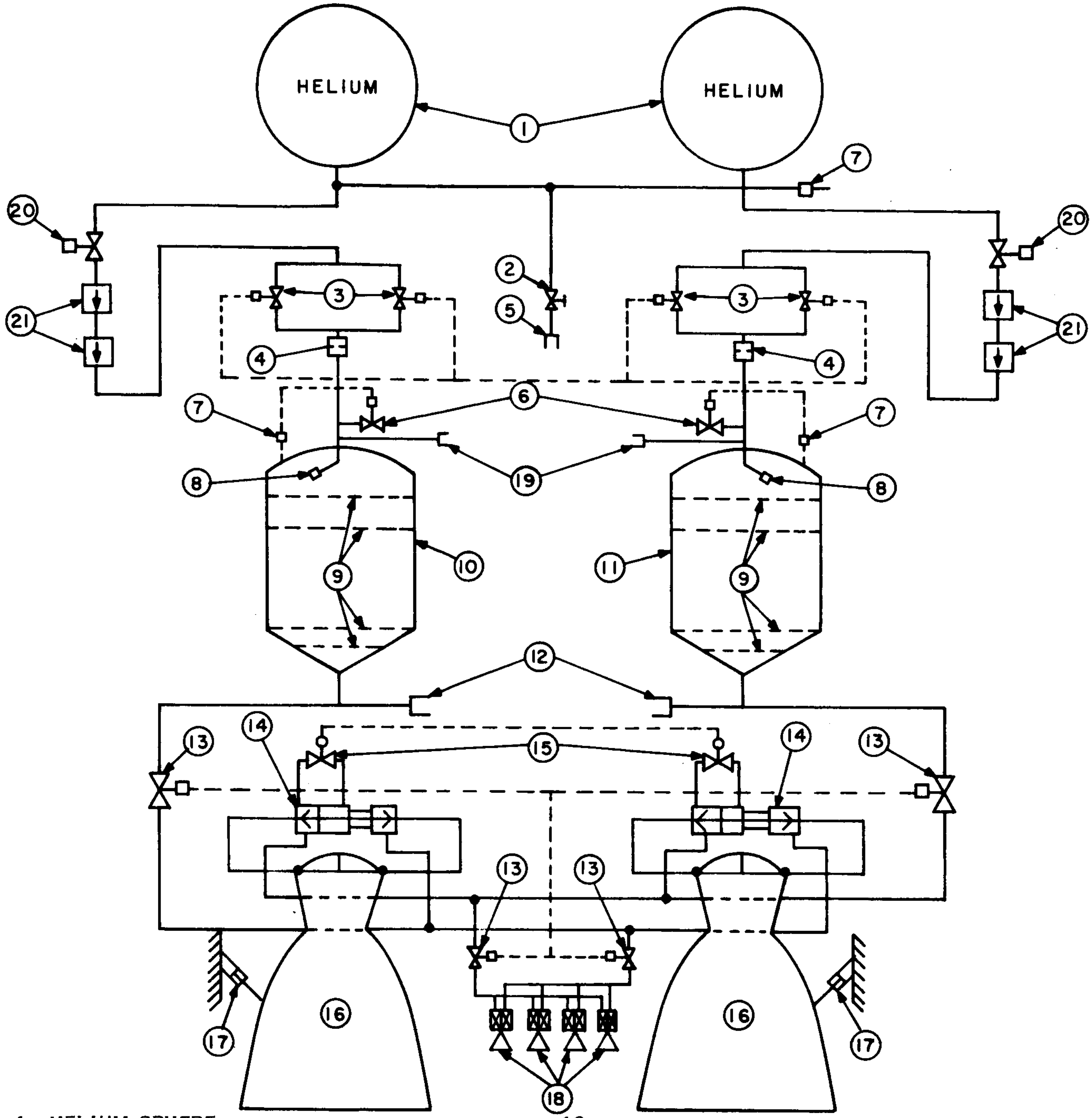
A schematic of the existing Transtage (configuration U-E) is shown in Figure 4.3-1. Configuration UE-3 (and MC-3) is shown in Figure 4.3-2, and configuration MC-1 is shown in Figure 4.3-3.

4.3.4 Subsystem Selection. Studies were conducted to select a single system from the foregoing configurations. The major areas investigated were probability of mission success (reliability), cost, contributions to subsequent missions, and performance and weight characteristics. These areas have been listed in decreasing order of importance with the first three being the most significant.

All system comparisons were based on the general requirements of a 200-meters/sec midcourse ΔV, 2.2 km/sec retro ΔV, and 100 meters/sec orbit adjust ΔV. The midcourse correction accuracy was taken as 1.0 $\pm$0.07 meter/sec with a desired goal of 0.1 $\pm$0.007 meters/sec. The 15,000-pound weight limitation only affected the 1975/77 systems and it was assumed to result in a restriction to retro ΔV capabilities only (MC/OA required stayed constant).

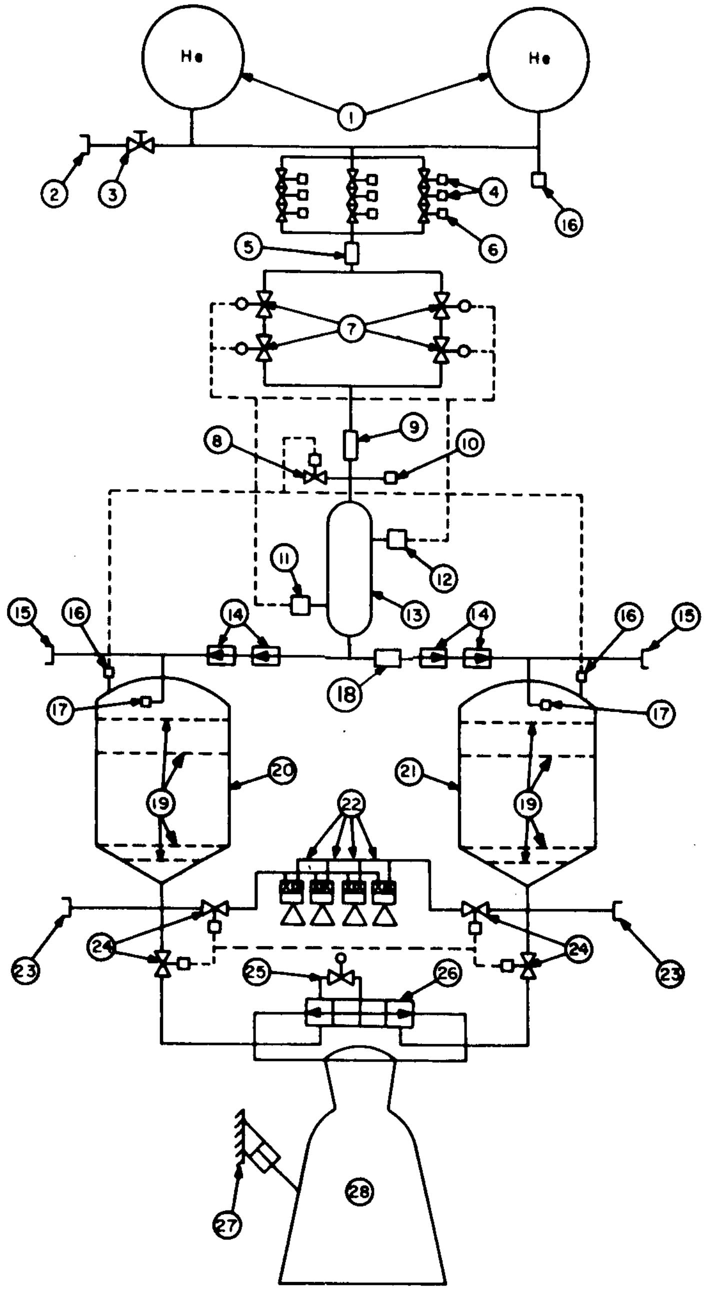
Each of the major areas are discussed in the following paragraphs.

4.3.4.1 Probability of Mission Success. Probability of mission success, or reliability, was considered to be the prime criteria for system selection. Since the mission duty cycles for the current Transtage applications, i.e., earth orbit insertion of satellites, are greatly



1. HELIUM SPHERE
2. HELIUM FILL SHUT-OFF HAND VALVE
3. ORDNANCE VALVES, NORMALLY CLOSED
4. HELIUM FLOW CONTROL ORIFICE
5. HELIUM SPHERE FILL CONNECTOR AND CAP
6. RELIEF VALVE, ORDNANCE OPERATED
7. READINESS MONITOR AND GROUND HANDLING SWITCH
8. HELIUM REFLECTOR
9. PROPELLANT ORIENTATION AND SLOSH CONTROL SCREENS
10. FUEL TANK
11. OXIDIZER TANK
12. PROPELLANT TANK FILL AND DRAIN CONNECTOR AND CAP
13. PROPELLANT PREVALVE, ELECTRIC MOTOR OPERATED
14. BIPROPELLANT VALVE, FUEL PRESSURE ACTIVATED
15. PILOT VALVE, SOLENOID OPERATED
16. THRUST CHAMBER
17. ENGINE GIMBLE ACTUATOR, HYDRAULIC
18. VERNIER ENGINES, FIXED
19. PROPELLANT TANK PRESSURIZATION AND VENT CONNECTOR AND CAP
20. ORDNANCE VALVES — NORMALLY OPEN
21. CHECK VALVES

Figure 4.3-2. Transtage Configuration U-E3 (Modified Pressurization Subsystem)



1. HELIUM SPHERE
2. HELIUM SPHERE FILL CONNECTOR AND CAP
3. HELIUM FILL SHUT-OFF HAND VALVE
4. ORDNANCE VALVE, NORMALLY OPEN
5. FILTER
6. ORDNANCE VALVE, NORMALLY CLOSED
7. SOLENOID VALVES
8. RELIEF VALVE, ORDNANCE OPERATED
9. FILTER
10. BLEED PORT
11. CHECKOUT AND PREFLIGHT PRESSURE SWITCH
12. FLIGHT PRESSURE SWITCH
13. GAS ACCUMULATOR
14. CHECK VALVES
15. PROPELLANT TANK PRESSURIZATION AND VENT CONNECTOR AND CAP
16. READINESS MONITOR AND GROUND HANDLING PRESSURE SWITCH
17. HELIUM DEFLECTOR
18. PRESSURE BALANCING ORIFICE
19. PROPELLANT ORIENTATION AND SLOSH CONTROL SCREENS
20. FUEL TANK
21. OXIDIZER TANK
22. VERNIER ENGINES, GIMBALED
23. PROPELLANT TANK FILL AND DRAIN CONNECTOR AND CAP
24. PROPELLANT PREVALVES, ELECTRIC MOTOR OPERATED
25. PILOT VALVE, SOLENOID OPERATED
26. BIPROPELLANT VALVE, FUEL PRESSURE ACTIVATED
27. ENGINE GIMBAL ACTUATOR, ELECTRIC
28. THRUST CHAMBER

Figure 4.3-3. Transtage Configuration MC-1 (Modified Propellant Tanks and Single Engine) different from the anticipated VOYAGER duty cycle, data for conducting a meaningful and quantitative reliability analysis is not available. Consequently, a qualitative analysis was conducted by using identifiable modes of failure as the parameters. The configurations U-E, U-E3 and MC-1 were evaluated to determine the potential causes for mission failure. The prime areas considered were valve malfunctions, tankage failures, leakage, and sequencing error.

Configuration U-E3, shown in Figure 4.3-2, incorporating a blow-down pressurization system, represents the most simplified and inherently reliable combination of pressurization system valving. It also completely eliminates the possibility of the inadvertent cross-mixing of the hypergolic propellants prior to the retro maneuver. The leakage potential of the helium system is greatly reduced through the use of low pressure (800 psia) gas storage, as compared to the 4500-psia storage pressure required for either U-E or MC-1. Furthermore, the blow-down technique with a large portion of the gas stored in the propellant tanks minimizes the possibility of total mission failure due to valve malfunction or sequencing error in the pressurization system. Some type of degraded missions will always be available. A manufacturing modification to the U-E3 propellant tanks will be required to accommodate the 2.2-to-1 safety factor required by the blow-down approach.

4.3.4.2 Cost. A detailed cost comparison of the competitive Transtage configurations was not conducted due to the lack of detailed data. However, the existing Transtage configuration, U-E, will certainly require the least development effort. Conversely, the two-fuel tank, single-chamber configuration, MC-1, will be most costly considering the requirement for major structural changes. The U-E3 blow-down configuration, with tankage and pressurization system modifications, would fall about midway between the cost extremes. The variation in cost across the range from U-E to MC-1 would probably not exceed 10%.

4.3.4.3 Contributions to Subsequent Missions. Contributions to subsequent missions were considered to be the ΔV capability for the 1975/77 VOYAGER Mission. All four systems have been sized for the 1975/77 VOYAGER Mission, but the two-fuel tank configuration, MC-1, has a definite edge due to its light weight and, thus, larger propellant load capability.

4.3.4.4 Performance and Weight Characteristics. The results of this study are presented in Table 4.3-2.

4.3.4.5 Conclusions. The blow-down pressurization system (U-E3 or MC-3) was rated superior to either of the other two design configurations based chiefly on the primary selection criterion, reliability. The simplified pressurization system, with minimal valving requirements, enhances the probability for attaining a highly reliable, leak-tight system.

As related to the criteria of cost and subsequent mission capability, the blow-down system, U-E3, suffers only slightly in comparison with the alternative configurations. In the inter-related areas of weight and performance, configurations U-E, U-E3, and MC-3 show little variation (a maximum spread of 0.050 km/sec in retro due to a weight difference of 293 pounds). Configuration MC-1 shows a retro ΔV 0.2 km/sec over the next system, which might be significant. However, this single factor was not considered to be decisive in the system selection.

TABLE 4.3-2. WEIGHT AND PERFORMANCE SUMMARY

Configuration	Dry Weight (lb)	Burnout (lb)	Retro Propellant Weight (lb)	MC&OA Propellant Weight (lb)	71/73 Total Weight (lb)	Retro I_{sp} (sec)	MCOA I_{sp} (sec)	75/77 $\Delta \cdot V_5$ (kmps)
1. U-E Existing Transtage with minimum modification	2549	2764	9315	1645	13,724	305	285	1.25
2. U-E3 Blow-down pressurization with heavy gage propellant tanks of present length	2386	2601	9400	1604	13,605	300	285	1.28
3. MC-3 Blow-down pressurization with heavy gage propellant tanks (shortened)	2306	2521	9300	1585	13,406	300	285	1.30
4. MC-1 2 Fuel tanks with existing pressurization system, one main engine	1743	1951	8570	1402	11,923	305	285	1.50

4.3.5 Operational Aspects of Selected Configuration

4.3.5.1 Functional Description. In the liftoff condition, the propellant tanks are pressurized to 170 psia. The helium spheres, isolated from the propellant tanks by the ordnance valves, are at 800 psia. All propellant valves and prevalves are closed.

When making the midcourse corrections, all velocity changes greater than 10 meters/sec will be made with the main engines. (All corrections less than 10 meters/sec will be made with the four vernier thrust chambers. All these maneuvers (200 meters/sec max.) are made without the addition of pressurization gas to the propellant tanks. The final pressure in the propellant tank (prior to retro) is 150 psia. To make a correction first requires the selection of the appropriate system, i.e., vernier or main thrust chambers. The prevalves in the selected system are then commanded open. The time between this operation and fire command is not critical but it should be minimized to reduce the possibility of leakage. The fire command is given which will effect the opening of the solenoid valves on the vernier thrust chamber or, depending upon mode of operation, the bipropellant valves on the main chamber. These valves allow propellants to flow into the chamber, where they ignite hypergolically. Burning continues until the shutdown signal is received from the vehicle. This signal causes the solenoid valves (bipropellant valves) to close, terminating flow and combustion. The prevalves are then closed, sealing the system.

The retro velocity (2.2 km/sec) is achieved by using the main engines. The operating sequence is the same as the preceding sequence, except the ordnance-operated isolation valves in the pressurization system are opened with the engine start signal. Propellant tank pressure rises to a maximum of 180 psia and decays during the firing to approximately 160 psia. The resulting chamber pressure decreases from 115 psia to 100 psia, which is within the present design limits of 120 to 90 psia.

The vernier thrust chambers are used to make all orbit-adjust velocity changes. The operating sequence is as described above. After the last maneuver, the normally open ordnance valve in the pressurant lines are fired closed, isolating the pressurant section of the system. The prevalves will provide sufficient isolation in the propellant lines.

4.3.5.2 Performance

4.3.5.2.1 Retro Velocity Accuracy. The engine impulse uncertainty from the main engines is less than ±780 lb-sec.

4.3.5.2.2 Midcourse and Orbit-Adjust Velocity Changes and Accuracy. The minimum controllable impulse available from the main engines is about 16,000 lb-sec. This corresponds to about 9.0 meters/sec for midcourse and 20.0 meters/sec for orbit adjust velocity changes. The uncertainty would be ±0.43 meters/sec for midcourse and ±1.0 meters/sec for orbit-adjust velocity changes. This data provides the criteria for vernier engine operation. The main engines could be used for all midcourse velocity changes greater than 10 meters/sec and the vernier engines for lesser velocity changes, since they will probably be small and would total less than 50 seconds burn from four 45-pound thrust vernier engines. The Transtage attitude control engines have been tentatively selected for use as the vernier engines. These are flight-qualified ablative chamber engines rated at 45 pounds of thrust with an operating life of 600 seconds. Four of these engines have minimum impulse and impulse accuracy well within the requirements of the VOYAGER Mission.

4.3.5.2.3 Weight Summary. A summary of the Transtage configurations dry and burnout weights is given in Table 4.3-2. The following is a summary of the trapped propellant and inert weights:

• Maximum outage at 1%, lb	100
• Trapped and unusable propellant, lb	50
• Pressurant gas, lb	32
• Propellant vapor, lb	48
• Ablative material consumed, lb	15

4.3.5.2.4 Growth Potential. The growth potential of the selected system is limited to 1594 pounds of additional propellant. This limit is imposed by the maximum propulsion module weight of 15,000 pounds. The resulting 1975/77 maximum retro velocity is 1.30 km/sec. If the weight restriction were removed, the existing Transtage would perform a retro velocity of 2.0 km/sec, but this would require a propulsion module weight of nearly 25,000 pounds. High-energy propellants are a possibility for performance improvement, but this would require the development of a new engine.

4.3.5.2.5 Packaging Envelope. The Transtage, as a propulsion module, packages within a 120-inch diameter ring frame. Its total length is 195 inches as shown in Figure 4.3-4.

4.3.6 Development Aspects of Selected Configuration. Following are discussions of those aspects of the selected configuration which will affect the development effort required for the system, from the standpoints of hardware modification, schedule risk, and reliability.

4.3.6.1 Mission and Space Storage Compatibility

4.3.6.1.1 Leakage Control. Leakage minimization would be given primary consideration during vehicle development. The design would provide for welding or brazing of all plumbing, with mechanical joints only at the connection of major subassemblies. The blow-down pressurization system would be compatible with this concept, since it has no components that require replacement.

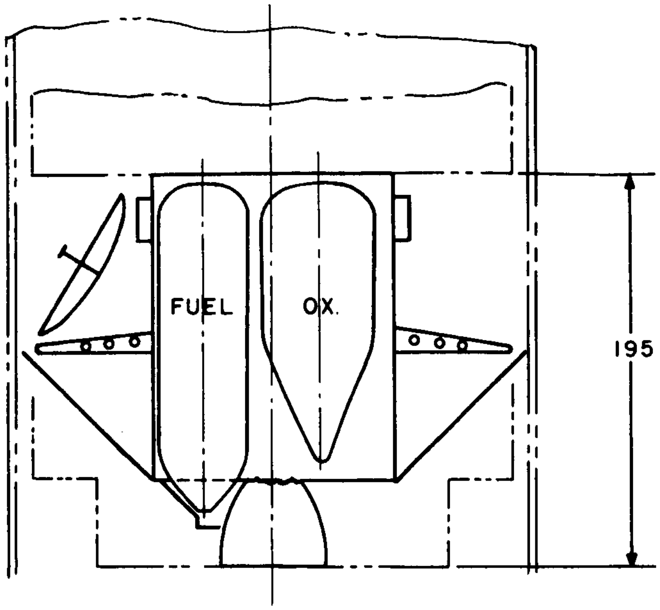


Figure 4.3-4. Transtage Configuration Packaging Envelope

The propellant feed line prevalves have the primary function of leak prevention through the engine valves. They will be a rotating poppet valve with two-stage linkage. The first stage would be used to rotate the valve into a closed position, with the second stage applying high mechanical advantage to assure positive sealing. The most probable drive mechanism is an electric motor. When the engines are inoperative, the lines between the prevalves and thrust chamber valves would be evacuated by positive propellant bleed. This will prevent freezing caused by uncontrolled leakage.

The leak compatibility of seals in the hydraulic actuation section will have to be further investigated. The motor-pump is presently encapsulated and pressurized and is, therefore, no problem. The seals of the actuator piston are the most probable source of leakage, but redundant seals or a pressurized boot over the actuator rods would prevent seal drying and provide seal lubrication, thereby reducing leakage. If necessary, the existing reservoir-accumulator volume can be increased to accommodate expected leakage.

4.3.6.1.2 Propellant Orientation. Propellant acquisition will be provided for the main engine and the vernier system by maintaining propellants in the tank bottoms at all times. This condition is made practical by the VOYAGER Mission propellant consumption schedule. At shutdown of the final launch vehicle stage, propellants will be bottomed and tanks will be approximately 50% full. During the transfer to Mars, midcourse corrections may reduce the propellant load to about 45% of tank capacity. This variation is predictable and the associated location of the liquid level during coast can be specified if the propellants are maintained in the bottomed condition. Following insertion into Mars orbit and during the orbit adjustment period, maximum and minimum propellant levels can again be located. For this coast condition, remaining propellant can vary from about 2% of tank capacity to some minimum usable level near the tank bottoms.

With this information, it is possible to limit propellant disorientation with special screen assemblies located in the tank areas corresponding to the two coast phases. (See Figure 4.3-5.) Fine mesh metal screens spanning the tank diameters will be attached to the tank walls a few inches below the lowest fluid level anticipated for each coast period. These screens will accomplish the basic purpose of suppressing surges created by shutdown transients which tend to disturb the propellant orientation. The submerged position of the screen will ensure that any liquid flowing forward will be replaced by liquid from above the screen. Proper placement of the screen will permit such circulation of liquid without admitting gas or vapor into the tank bottom region for the magnitude and nature of the disturbances expected. Mesh size of these lower screens will be selected to provide maximum surge damping while permitting free flow of liquid during periods of engine operation.

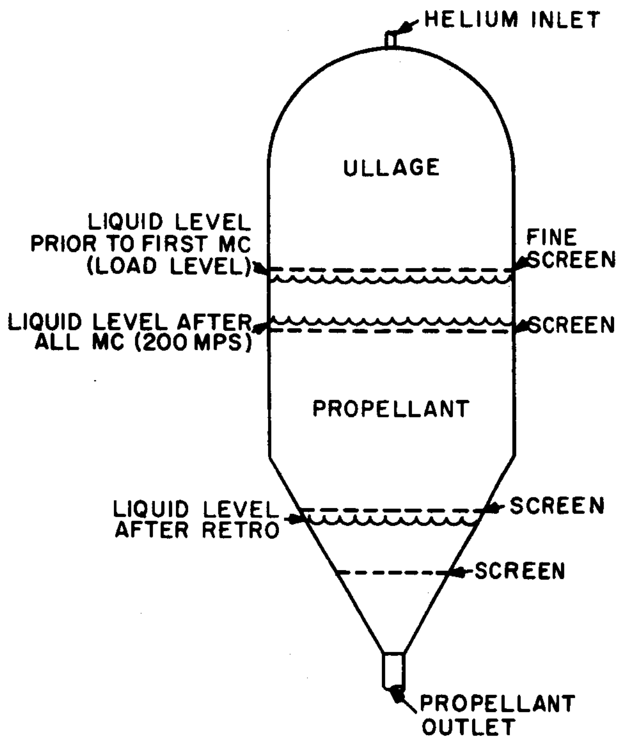


Figure 4.3-5. Schematic of Propellant Acquisition System

A second set of screens will be installed a few inches above the highest fluid level anticipated for each coast phase. These screens will be of finer mesh, offering maximum resistance to passage of liquid while permitting free flow of pressurant gas. The liquid/gas interface will be trapped in the volume between lower and upper screens. This condition facilitates suppression of propellant slosh with simple transverse baffles which may be incorporated with the screens into composite assemblies.

Tank top volumes will be filled with pressurant gas and propellant vapor. During coast conditions a thin film of liquid will wet the tank walls and provide an efficient barrier between the helium and potential leakage areas. A secondary advantage of segregating the tank gases in a predictable location is compatibility with simple methods of accomplishing tank venting if required.

The current propellant traps and baffles can be removed from the tank bottoms. Antivortex baffles will be incorporated with the aft screen assemblies for the VOYAGER Mission application.

4.3.6.1.3 Stress Corrosion of Titanium Tanks. Work on the NASA Apollo Program has uncovered an incompatibility between nitrogen tetroxide and titanium pressure vessels. It appears to be a stress level-time-temperature problem compounded by the presence of free chlorine in the propellant. The most promising areas being investigated are: (a) controlling the free chlorine in the propellant, (b) coating the tanks with titanium oxide or teflon, and (c) the addition of corrosion inhibitors to the propellant. It is reasonable to assume that the present concentrated efforts throughout the aerospace industry will produce a solution before the start of the VOYAGER Program. Thus, it is concluded that the Transtage propellant tanks will be fabricated from titanium.

4.3.6.2 Technical Risk and Development Time. Since the existing Transtage is operational, there should be no risk involved with the basic system. The proposed modifications are all state of the art; therefore, no major problem is anticipated in development. The following is a brief summary of the major modifications.

a. Pressurization System

The only components on this system are the ordnance valves of which there are several qualified units available. The basic system can be proven in ground testing with hardware available at the Martin Company. Therefore, it would not present a problem with respect to development time. As a result of pressurizing the propellant tank to operating pressure on the ground, these tanks will be redesigned to a factor of safety of 2.2. This is possible without major redesign of tooling, since the tanks are fabricated from approximately 1-inch thick forgings and machined to the required thickness. Since the helium spheres will contain lower pressures, they will be machined down to the thickness required for a 2.2 factor of safety.

The present propellant tank, helium sphere, and engine support structures will be retained. It will be considerably overdesigned due to the lighter propellant loads and sphere weights, but this should add to the overall vehicle reliability.

b. Propellant Line Prevalves

A prevalve of suitable design is presently under development by the Martin Company. It is now being tested in long-term storage and should be available in time to support the VOYAGER Program.

c. Propellant Acquisition Screens

An industry-wide search for passive zero-gravity propellant control has produced considerable data on screens. This data indicates that screens are a feasible method of propellant control. Since the screen application for the VOYAGER Mission is of a holding nature rather than collecting nature, the basic function can be proven in a gravity field. Therefore, test verification is provided.

d. Vernier Thrust Chamber Assemblies

There are several flight qualified bipropellant assemblies in the 50- to 100-pound thrust class. (e.g., Thiokol, RMD, C-1 for NASA; Marquardt LEM RCS; and the Rocketdyne Transtage ACS chamber). Any of these engines can be used as is (i.e., without modification to the present propellant valves), since they will be backed up by prevalves in the propellant lines.

4.4 LEM Descent Propulsion

4.4.1 System Discussion

4.4.1.1 Existing LEMDE System. The Lunar Excursion Module Descent (LEMDE) Propulsion System is a complete bipropellant propulsion system consisting of propellant storage, pressurization and feed components all housed within a cruciform structure which forms a basic part of the LEM Space Vehicle structure. An ablative thrust chamber with a radiatively-cooled skirt is combined with a unique variable area injector/throttle valve to permit operation from 10,500 pounds thrust down to 1050 pounds thrust. The propellant is a 50-50 mixture of hydrazine and unsymmetrical dimethylhydrazine as the fuel and nitrogen tetroxide (N_2O_4) as the oxidizer. Descriptions of this system are presented in detail in the JPL "Design Data for Candidate VOYAGER Spacecraft Propulsion Systems" dated 12 November 1965. The module is under development by the Grumman Aircraft Engineering Corporation and is scheduled to be qualified in 1967.

4.4.1.2 Application to VOYAGER. First estimates of retropropulsion requirements for the 1971 VOYAGER Spacecraft indicate a need for a total impulse of 2.5 to 3 million pound-seconds. If a burn time of 600 seconds is assumed as a reasonable duration for a bipropellant ablative thrust chamber, the minimum thrust for VOYAGER retropropulsion would be 5000 pounds.

A 10,000-pound thrust engine burning for 300 seconds would, therefore, satisfy immediate requirements and allow a large margin of safety for future growth. At the same time 10,000-pounds of thrust would impose a maximum acceleration during orbit insertion only slightly in excess of one g. Thus, the LEMDE thrust level is well suited for VOYAGER retropropulsion requirements.

A major LEMDE system change necessitated by VOYAGER requirements would be the pressurization system. The present supercritical helium storage system is not considered state-of-the-art for the extended time in space as required by the VOYAGER Mission. By returning to the "original" LEMDE high pressure helium storage system, a satisfactory system could be achieved. (The original system is patterned after the conventional gas-regulated method of pressurant management. Quadregulators are used to reduce the pressure from storage to working conditions. (See Figure 4.4-1.) If the added weight of this system were to become excessive, heat exchangers or partial blowdown systems could be considered to minimize pressurization system weight. However, for initial comparisons, helium gas stored at 3500 psig was assumed.

A second modification would be the need for new gimbal actuators to meet the higher response rates required by VOYAGER.

4.4.1.3 Alternate LEMDE Configurations. An initial evaluation of the existing LEMDE system shows two areas which, although necessary for LEMDE, are not required for VOYAGER. First is the large diameter of the basic LEM structure, approximately 166 inches. This is nearly four feet greater than the desired maximum of 120 inches for the propulsion system.

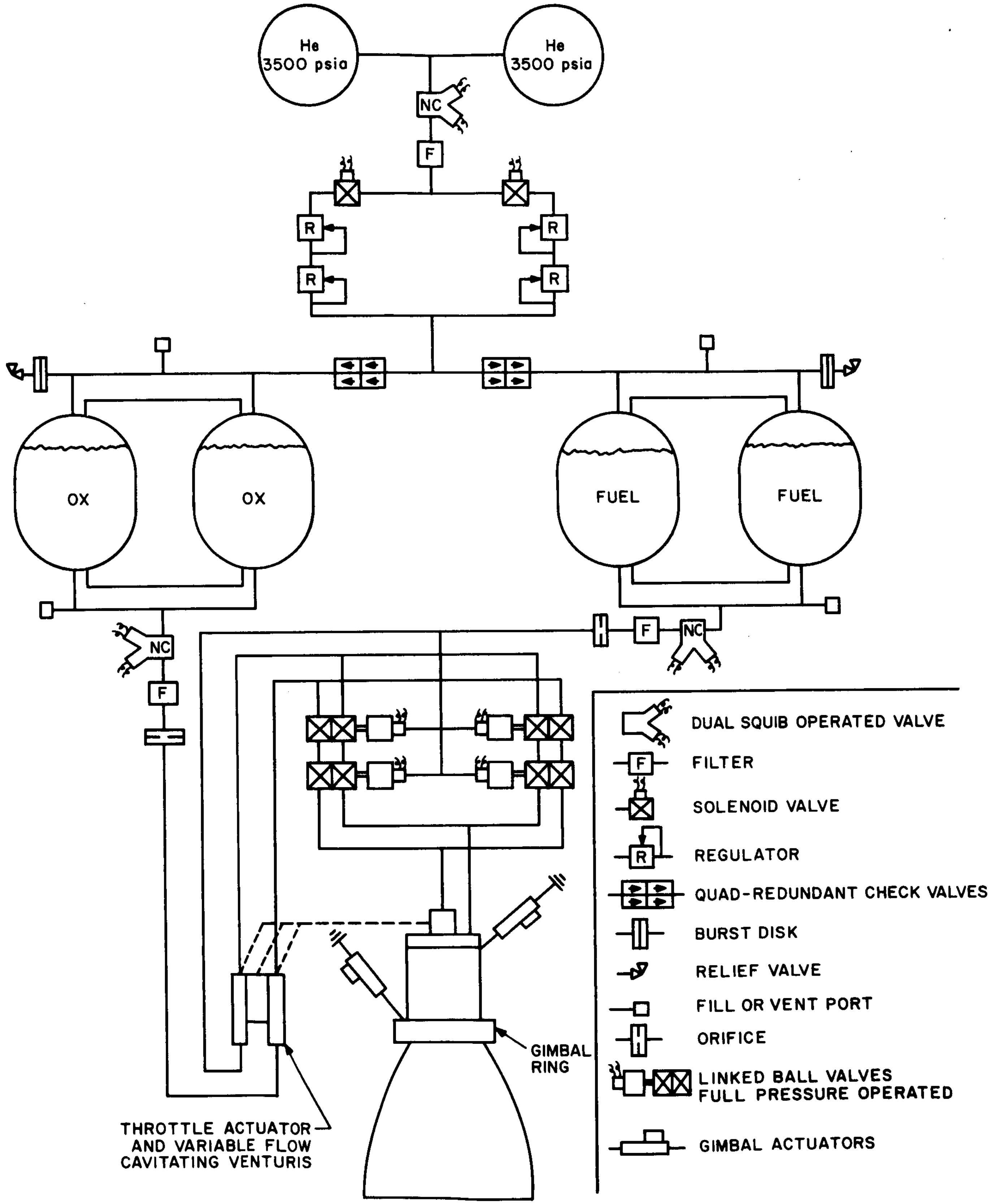


Figure 4.4-1. Model LEMDE Schematic

Second is the weight of the LEMDE structure. Designed for severe landing stresses, it far exceeds the design load requirements of VOYAGER. This combination of size and excess structural weight suggests that a repackaging of the LEMDE components could result in a better package for VOYAGER without sacrificing the proven reliability of these components.

As a further packaging refinement, a change in propellant tank length was considered. The existing LEMDE tanks will hold approximately 18,000 pounds of propellant. Maximum VOYAGER propulsion system weight is limited to 15,000 pounds, of which about 12,000 pounds would be propellant. Thus, the present tanks hold an excess of nearly 6000 pounds of propellant. By using the existing hemispherical end tanks it should be possible to lengthen or shorten the cylindrical section without appreciably affecting tooling costs. Therefore, a ground rule for repackaging was established such that tank size on any LEMDE system, other than the basic configuration, should result in a loaded propulsion system weight of 15,000 pounds for the 1975/77 Missions. This results in only a small weight penalty (about 20 pounds in 1971/73).

A candidate method for repackaging is to retain the four tank system for symmetry and cg control. However, the tanks are moved into a minimum diameter and the thrust chamber lowered as required. This permits maximum utilization of existing LEM packaging techniques and plumbing configurations.

A second arrangement is to use only two tanks and offset them from the centerline by the ratio of oxidizer weight to fuel weight (1.6). This will maintain the cg on the vehicle centerline as the propellant is used.

No other configurations appeared to offer advantages over those discussed above. Thus, the three configurations which were evaluated are:

- LEMDE as is (high pressure stored helium)
- Modified LEMDE, 4 tanks
- Modified LEMDE, 2 tanks

A system schematic which will apply to any of these systems (except for the number of tanks) is shown in Figure 4.4-1. One addition to the LEMDE system which would be proposed for the VOYAGER system is the normally-closed explosive valve in each of the main propellant lines for positive sealing of propellant prior to launch.

4.4.2 Performance

4.4.2.1 MC/OA Capabilities. The previous discussion of LEMDE propulsion capabilities was limited to retropropulsion requirements. Since the LEMDE thrust chamber has the ability to throttle down to 1050 pounds of thrust, this presents the possibility that a single engine could perform all midcourse, orbit-insertion and orbit-adjust maneuvers. Lack of precise information on the minimum impulse bit and accuracy of cut-off of the LEMDE thrust chamber make it difficult to accurately determine minimum ΔV's for the different maneuvers. However, based on the characteristics of similar chambers (not specifically designed for rapid shutdown) it should be possible to obtain a minimum impulse bit of 500 pound-second when operating at the 1050-pound thrust level. For 1971 VOYAGER weights this means a minimum midcourse ΔV of about 0.25 meter/second and a minimum orbit adjust correction of about 1.0 meter/second. Since the required midcourse correction is 1.0 meter/second with a desired capability of 0.1 meter/second, the LEMDE engine should be capable of meeting minimum ΔV requirements and approaching the desired ΔV values.

Accuracy of cutoff for minimum impulse maneuvers can only be assumed to be within the desired ± 7% since this value is not unreasonable for most thrust chamber assemblies. If the existing system should prove unable to meet such accuracies, modification of the shut-off valve actuation system could probably bring the response and accuracy within required limits. Thus, by reason of the throttling (or, more exactly, two-thrust level operation) feature, any of the LEMDE propulsion systems could perform all mission propulsion maneuvers.

To evaluate LEMDE from every aspect, the selected systems were also examined parametrically in conjunction with the following MC/OA systems:

- Separate monopropellant system
- Separate bipropellant system
- Four small thrust chambers operating from the main tanks

A weight breakdown for a typical LEMDE configuration for 1971/73 is shown in Table 4.4-1. It may be noted that the retropropulsion system inert weight was not adjusted for varying tankage requirements. However, this effect is small and the change in weights would vary

TABLE 4.4-1. WEIGHT BREAKDOWN FOR DIFFERENT MC/OA SYSTEMS 1971/1973 MISSION

	MC/OA Inert Propulsion		Retropropulsion Inert Weight (lb)	Retropropulsion Weight (lb)	Total Propulsion Weight (lb)
	Weight (lb)	Weight (lb)			
LEMDE Alone	---	1468	2343	8978	12789
LEMDE + Monopropellant System	522	2086	2343	9676	14627
LEMDE + Bipropellant System	379	1518	2343	9408	13648
LEMDE + Thrust Chamber Only	77	1462	2343	9602	12944

only slightly. Completely separate systems, whether monopropellant or bipropellant, fall within the 15,000-pound limit, but from the total weight standpoint, they do penalize the system on a growth basis. The separate thrust chambers operating from the main tanks impose a much smaller weight penalty and, at the same time, offer several back-up modes of operation that might be attractive. These include engine-out capability if the four MC/OA thrust chambers are arranged in symmetrical pairs and a back-up mode for MC/OA by use of the main thrust chamber. The primary advantage of the small thrust chamber configuration is that it allows the main retropropulsion thrust chamber to be sealed off until it is prepared for the orbit insertion maneuver.

Reliability estimates of the four types of systems show a slight decrease in overall mission reliability from that of the LEMDE system alone when either the separate monopropellant or bipropellant MC/OA systems are used. The four separate MC/OA thrust chamber configuration operating from the main tanks shows a slight increase in reliability over the single system. Differences in reliability are so small that there is no justification on this basis for a separate MC/OA system in conjunction with any LEMDE configuration. All further studies, therefore, assumed that the LEMDE engine (in any configuration) performed all propulsion maneuvers.

4.4.2.2 Weight. Estimated inert weight breakdown for each of the LEMDE systems is given in Table 4.4-2. Each of these inert weights was assumed for the 1971/73 Mission

TABLE 4.4-2. ESTIMATED INERT WEIGHT BREAKDOWN-LEMDE SYSTEMS

	As Is			Modified 4 Tanks			Modified 2 Tanks		
	No.	Unit Wt (lb)	Total Wt (lb)	No.	Unit Wt (lb)	Total Wt (lb)	No.	Unit Wt (lb)	Total Wt (lb)
Thrust Chamber (Complete)	1	399	399	1	399	399	1	399	399
Propellant Feed System									
Propellant Tank	4	118	472	4	94	376	2	203	406
Plumbing			57			57			38
Trapped Propellant			474			474			348
Pressurization System									
Pressurant Tanks	2	240	480	2	190	380	2	190	380
Plumbing		36	36		36	36			30
Helium		44	44		35	35			35
Miscellaneous Hardware			21			21			18
Structural Weight			1036			206			195
Total Propulsion Inert Weight			3019			1984			1849

TABLE 4.4-3. PROPULSION SYSTEM WEIGHTS FOR THE 1971/1973 MISSIONS

	W_{ir} (lb)	W_{pm} (lb)	W_{pr} (lb)	W_{pt} (lb)	W_{tot} (lb)	ΔV (mps)
LEMDE As Is	3019	1601	9700	11301	14320	2200
Modified 4 tanks	1984	1399	8532	9931	11915	2200
Modified 2 tanks	1849	1375	8412	9787	11636	2200

W_{ir} Inert Propulsion System Weight
W_{pm} MC/OA Propellant Weight
W_{pr} Retropropulsion Weight
W_{pt} Total Propellant Weight
W_{tot} Total Propulsion System Weight
ΔV Orbit Insertion Velocity Increment

and complete propulsion system weights computed. These weights are shown in Table 4.4-3. The table indicates that:

a. The existing LEMDE structure places the as-is total weight for propulsion very near the allocated upper limit and allows little margin for growth.

b. The weight differences between the modified systems are insignificant. Any one of the systems, however, is capable of meeting 1971/73 requirements.

4.4.2.3 ΔV Capabilities and Weights for 1975/1977. The influence of the inert propulsion system weight on 1975/77 orbit insertion velocity capability and weights is shown in Table 4.4-4. The effect of the existing LEMDE structural weight is again immediately apparent in the ΔV capability, about 200 meters/second less than either of the modified configurations.

TABLE 4.4-4. ORBIT INSERTION VELOCITY AND WEIGHTS - 1975/1977

	W_{ir} (lb)	W_{pm} (lb)	W_{pr} (lb)	W_{pt} (lb)	W_{tot} (lb)	ΔV (mps)
LEMDE As Is	3019	2447	9534	11981	15000	1330
Modified 4 tanks	1984	2400	10616	13016	15000	1521
Modified 2 tanks	1849	2395	10756	13151	15000	1546

4.4.2.4 Packaging Envelope. Major packaging dimensions for the three systems under consideration are presented in Figure 4.4-2. Figure 4.4-2A is the existing LEMDE configuration. Figures 4.4-2B and C show the repackaged four- and two-tank configurations. These latter envelopes are based upon tanks sized to give a propulsion loaded weight of 15,000 pounds. Configuration B represents the minimum modification to the basic LEMDE engine which permits packaging within the desired 120-inch diameter. Overall length must increase from 106 inches to 173 inches, because of the new thrust chamber location.

Configuration C shows the packaging problems involved with a two-tank system. To maintain the cg location on the thrust axis as propellant is used, the tanks must be offset by the ratio of oxidizer to fuel weight. To balance the inert weight, other components or subsystems must be packaged opposite the fuel tank. With the oxidizer tank tangent to the thrust centerline the major lateral dimension is 120 inches. However, there is no dimensional symmetry about the centerline and the package must violate the 120-inch diameter allowance if the thrust axis is to coincide with the vehicle axis.

By moving the oxidizer tank partially across the thrust axis a point is reached where the fuel tank may be mounted tangent to the oxidizer tank. This reduces the lateral dimension to 102 inches, but induces lateral motion of the cg during burn periods. Although predictable, this cg motion is not desirable.

Length of the two-tank package as shown is 185 inches. This dimension could be greatly reduced by separation of the tanks, but the major lateral dimension approaches, or exceeds, that of the basic LEMDE package.

From a propulsion standpoint the two-tank system has the highest reliability because of the reduced component and plumbing requirements. However, the packaging and vehicle integration problems are severe.

4.4.3 Propellant Control

4.4.3.1 Sloshing. One of the ground rules for LEMDE modifications was that the basic tank configuration would be maintained. Sloshing in LEMDE tanks is controlled by baffle assemblies in each tank. These baffles (including antivortexing baffles) would be included in any of the modified systems. Preliminary analysis indicates that the present system is adequate during propulsion maneuvers for suppressing slosh modes which might affect the autopilot. Of greater concern are sloshing interactions with the Attitude Control System and, with four-tank systems, gross movement between parallel tanks during cruise or orbiting periods. If control of propellant position within individual tanks is important, a method of using screens to restrain movement within small limits has been proposed by the Martin Company and appears to be quite feasible. The predictable usage of propellant during the various mission

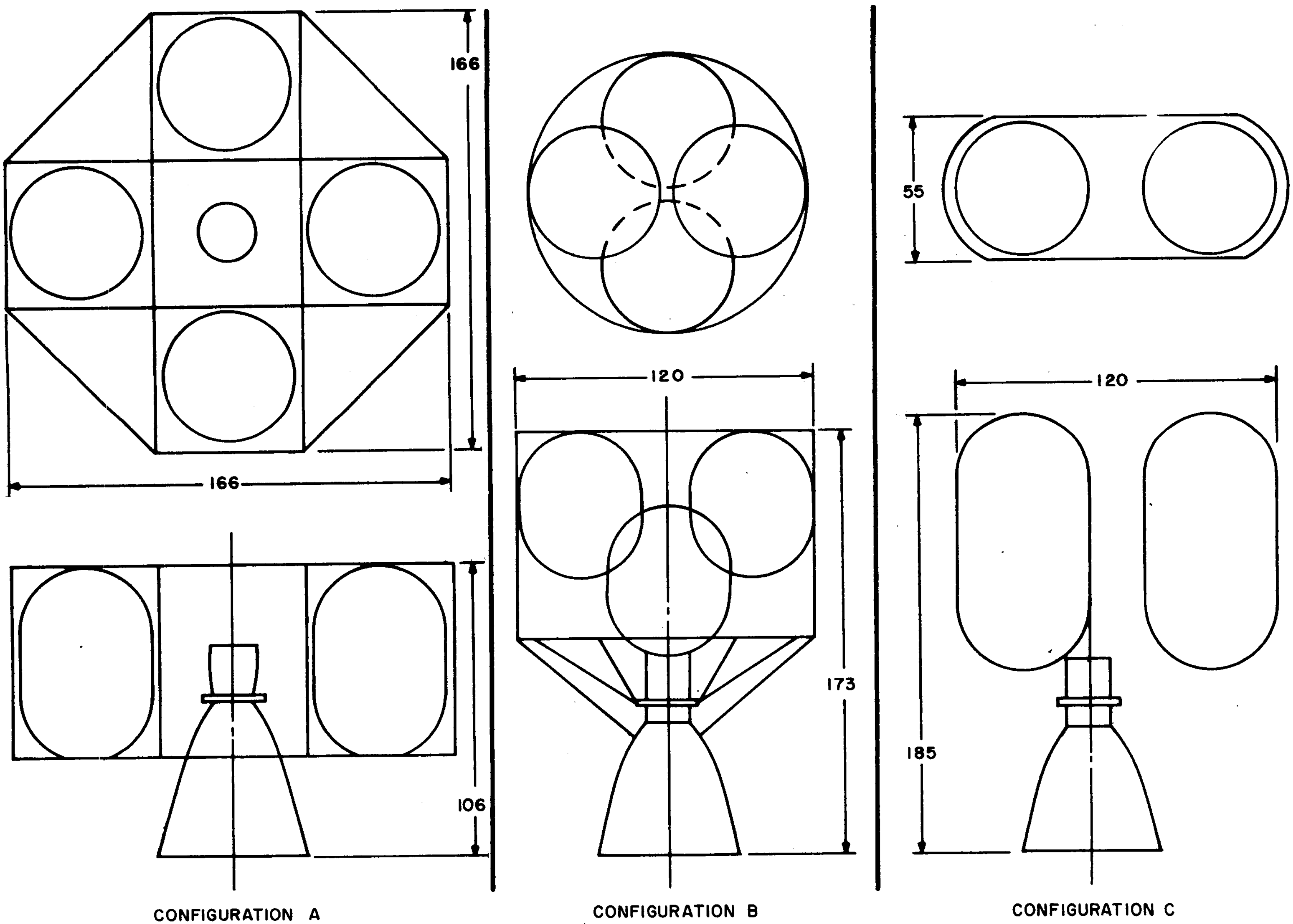


Figure 4.4-2. Packaging Dimensions for Three LEMDE Configurations

phases allows placing of these control screens to bracket possible liquid surface positions, thus preventing both sloshing and gross movement. Screens of this type could be used in either four- or two-tank systems. Although not proven state-of-the-art components, control screens fall within one of the major development areas required in bipropellant liquid space engines, i.e., propellant control. As such, continued effort would be recommended in this area, even with a relatively advanced system such as LEMDE.

4.4.3.2 Propellant Acquisition Under Zero-g Conditions. The present LEMDE propulsion system settles the main propellants by firing the Reaction Control System in the LEM Ascent Module. There is no provision within the basic descent module to perform this function. Thus, for VOYAGER application, new concepts or auxiliary systems must be employed. An analysis has shown that cold gas jets, even at the low thrust level of the Attitude Control System, can settle the main propellants with a gas usage of about three pounds per engine start, a value which is quite reasonable.

Existing programs on propellant acquisition with surface tension devices may well prove the capabilities of these concepts in the very near future. Such devices, coupled with the slosh control screens previously mentioned, could conceivably provide all the needed propellant control for the LEMDE propulsion system. No matter which LEMDE system is selected, development work in this critical area would be required.

4.4.3.3 Propellant Outage. Available figures for the existing LEMDE engine give propellant outage values (unusable propellant) varying from 400 to 650 pounds or about 3% of the total propellant weight. This is rather high, but is explainable in part because of the use of the tank crossover lines and the fact that the spherically-ended tanks do not lend themselves to minimization of propellant outage. From the propellant outage standpoint, the two-tank LEMDE engine would be lighter in weight than the four-tank systems because of the minimized plumbing and trap areas. This reduction in trapped propellant is the main factor that contributes to the slightly improved performance (and lowest total system weight).

4.4.4 Maneuvering Capabilities

4.4.4.1 Thrust Vector Control. The basic LEMDE thrust chamber incorporates a complete gimbal system mounted at the throat plane. Each of the systems under consideration would use this gimbaling capability for pitch and yaw control. As previously mentioned, desired response rates greater than those used in the present LEMDE system require a redesigned actuator. No problems with either the components or the overall engine are anticipated as a result of these requirements.

4.4.4.2 Roll Control. No method for providing roll control is incorporated in the basic LEMDE propulsion system. The magnitude of any roll forces generated in a liquid rocket engine are very small and can be accommodated by the cold gas jet system. If higher roll control torques should be required for some reason, the use of small gimbaled or hinged bipropellant thrust chambers operating from the main propellant tanks is a possible solution.

4.4.5 Overall Spacecraft Considerations

4.4.5.1 Sterilization. Sterilization procedures for bipropellant propulsion systems are not clearly defined at this time. If a sterilization requirement were to be placed on the LEMDE engine, it could involve a large effort. For example, if heat cycling were to be selected as the most practical method, it might require requalification of many engine components to meet the increased temperature-time conditions. Effectiveness of the treatment, chances for recontamination during loading and ground check-out and determination of other potential problem areas could involve significant effort. Sterilization, in any event, would affect each of the three configurations equally.

4.4.5.2 Magnetic Cleanliness. It is safe to assume that the LEMDE engine was designed without regard to the magnetic cleanliness requirements applicable to the VOYAGER Spacecraft. Thus, every component of the engine must be evaluated as to its magnetic properties (including all electrically operated components) and suitable redesign undertaken as required. As with sterilization, it is probable that many of the components would require requalification. Whereas sterilization is a possible requirement, magnetic cleanliness will be a definite consideration.

4.4.6 Long Life and Space Storage. The LEMDE engine is designed for an unknown ground life and for a relatively short life in space. It will require complete analysis and evaluation of all components to assure the inherent potential to meet long life prior to launch, and a capability to withstand the extended period in deep space required by VOYAGER. When these evaluations are complete and required changes identified, a program should be undertaken to establish that the engine can meet such requirements.

4.4.7 Reliability. Information was not available in sufficient detail to permit a complete reliability analysis of the LEMDE system. However, it can be assumed that because of the intended application of this stage, the reliability requirements that have been imposed on its design must be extremely high. It is expected, therefore, that this system could reliably perform the VOYAGER Mission if:

- No fundamental limitations exist to prevent extension of its space storage capability from the few days currently required
- Adequate means of propellant acquisition can be provided
- Motion of large quantities of liquids during the cruise phase can be controlled so as to prevent severe interaction with vehicle attitude control.

The last item is of most concern, and its role in affecting the choice of the preferred system is discussed in Section 2.0.

4.4.8 Development Risks

4.4.8.1 Technical. There is little doubt that the LEMDE propulsion module will be completely qualified in 1967. Problems, such as the present stress corrosion of titanium tanks with nitrogen tetroxide may require changes in tank material or increased quality control of the propellant, but this should have no significant effect on overall schedules. Further, qualification of the system to meet long term space storage should be a low risk program. However, two specific VOYAGER requirements already mentioned magnetic cleanliness and sterilization, could significantly affect schedule. If complete redesign is required, then technical risks in requalifying the system are most certainly involved. Of the three LEMDE configurations the least technical risk must be assigned to the existing as-is engine because of the effort already concentrated on its development.

4.4.8.2 Schedule. Schedule risks must, of necessity, include technical risk and potential interference with existing LEMDE plans and commitments. Lack of definitive information in either area makes a sound decision difficult. However, based on the VOYAGER Program schedule it is felt that any of the three configurations could meet the anticipated propulsion system schedules.

4.4.8.3 Cost. Cost information on LEMDE is nonexistent.

4.4.9 LEMDE System Recommendations. From the foregoing discussions there is not a strong differentiation between the three LEMDE configurations from the propulsion viewpoint alone. Any of the systems can meet the basic propulsion requirements. Either of the modified systems shows a small advantage over the as-is system derived primarily from the weight and 1975/77 performance characteristics. Thus, the selection of the particular LEMDE engine configuration for the VOYAGER application must be based on vehicle system considerations rather than the propulsion capabilities.

4.5 Alternate Approaches. In determining approaches to meet the VOYAGER propulsion requirements, the majority of activity was spent in studying, accumulating data and evaluating equipment that has been developed and qualified on other space programs and is, therefore, current state-of-the-art. Specifically, with respect to MC/OA propulsion, propulsion vendor study support was solicited from those companies that have developed components suitable for direct application. For example, Thiokol Chemical, Reaction Motors Division was requested to study a system based on the use of the NASA C-1 100-lbf "universal" thrust chamber. The Marquardt Corp. studied a system incorporating their 100-lbf thrust chamber used in the NASA LEM vehicle reaction control system.

However, to provide in depth coverage, two concepts were evaluated that, in a limited sense, either are not current state-of-the-art or within the guides set forth by JPL. While these concepts were not seriously considered during the selection process, they are of interest and have sufficient potential advantages that they should receive future consideration during later studies. The first of these concepts is the use of a beryllium thrust chamber as studied by Rocketdyne, and the second, a study by Aerojet General Liquid Rocket Operations, using four 2200-lbf thrust chambers with pair-out capability.

4.5.1 Beryllium Thrust Chamber. Considering the beryllium approach first, the following is extracted from the Rocketdyne study:

BERYLLIUM ENGINE OPERATING PRINCIPLE AND EXPERIENCE

Advanced technology programs for analytical and experimental evaluation of beryllium as a thrust chamber material have been under way at Rocketdyne for over a year. Beryllium, with its unique properties of high heat capacity, high thermal conductivity, and low weight, provides a method of conducting the heat (generated in the high-heat flux area of the nozzle) to a low-temperature surface of the combustion zone, which in turn is cooled by a boundary layer fuel film introduced through the injector. Utilizing this internally cooled, heat sink principle ("interegen" concept) of beryllium permits the design of a rocket engine assembly that is insensitive to mission duty cycle variance and exhibits unlimited life capability.

To date, rocket engines have been designed and tested at the 4-, 5-, and 100-pound thrust levels. Experience at the 100-pound thrust level includes over 87 tests with a cumulative hot-fire time in excess of 20, 000 seconds including a 10,000-second steady-state firing on a single engine with no loss of performance or damage. Consistent vacuum specific impulse values in excess of 295 seconds have been demonstrated at an expansion ratio of 40:1 and a mixture ratio of 16:1. In the most recent tests the specific impulse has been exceeding 300, and with an ϵ of 60:1 or greater, specific impulse should consistently exceed 305 seconds.

A contracted Rocketdyne program is scheduled to start within 1 month under the auspices of the Air Force Rocket Propulsion Laboratory to refine the 100-pound-thrust design. In addition, a company-funded 1000-pound-thrust rocket engine program has been initiated which is oriented to specifically support the VOYAGER program.

BERYLLIUM ENGINE DESIGN

The beryllium engine (Figure 1) is fabricated from a beryllium powder which is hot pressed into a shaped billet that approximates the desired contour of the chamber. This formed billet is then finish machined to the final dimensions. The outer wall is machined to conform to the results of a detailed heat transfer analysis based on accumulated test data. The area near the flange that is attached to the injector is machined to a minimum allowable cross-section area to minimize heat conduction into the injector. However, a section modulus capable of withstanding all vibration loads is retained. A metal K-seal is used to prevent combustion gas leakage between the injector and chamber. The K-seal is especially applicable for long durations in outer space, and provides an excellent thermal barrier because of its minimum contact area with the two flanges.

Analyses have indicated that the maximum outer-wall temperatures reached with the beryllium engine will approximate those attained on the 100-pound-thrust engine tests. A summary of those temperatures is shown below:

Location	Maximum Wall Temperature, F
Beginning of Contraction	970
Throat	1150
$\epsilon = 20$	1550
$\epsilon = 40$	1600

The radiation effect of these maximum temperatures can be controlled by insulating the engine. Since engine operation does not depend on radiation cooling, there will be no deleterious effect on chamber life. Additionally, through proper design and by taking radiation cooling into consideration, heat soakback can also be controlled.

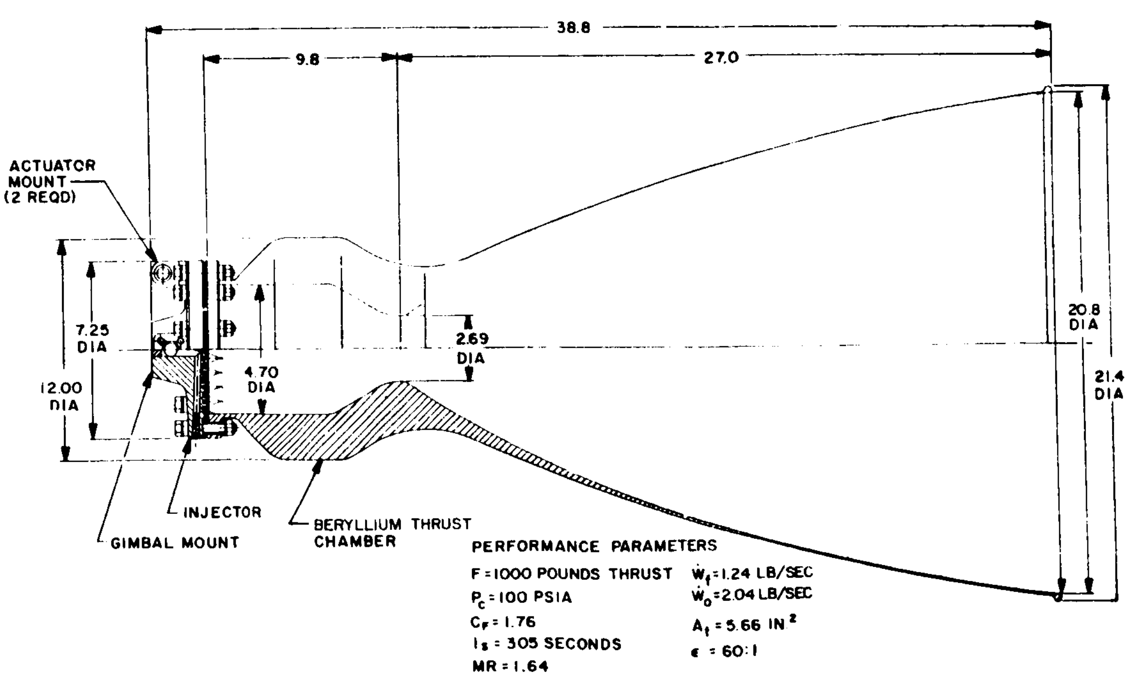


Figure 1. Beryllium Engine

This thrust chamber would be integrated with a propellant feed system that conforms to the design outlined in VC238FD101. Supported by this existing technology it would be possible to use 100-lbf thrust chambers in multiple chamber configurations or, as Rocketdyne proposes, continue development of their company funded 1000-lbf chamber which, as a single unit, would reform all propulsion functions.

4.5.2 Aerojet Study. The second approach was studied by Aerojet General. The following is extracted from their report:

I SYSTEM DESCRIPTION AND OPERATION

A. INTRODUCTION

The Aerojet-General AJ10-131 2200-lb thrust ablative bipropellant engines perform all midcourse and orbit adjustment ΔV requirements, including meeting the required minimum ΔV of 1 mps ± 0.07 mps accuracy. Retro would normally be accomplished with all four engines, although two engines could be used as a backup mode. Typical nominal engine durations are 20 seconds per MC, 297 seconds per retro and 4.5 seconds per orbit adjustment for a total of 395 seconds per engine. General design ground rules are given in Table I-1, and a weight summary is given in Table II-2

TABLE I-1. VOYAGER GENERAL PROPULSION SYSTEM DESIGN GROUND RULES

Mission				
Payload	3000-lb lander and 2500-lb spacecraft for 1971/1973			
Operation	MC	Retro	OA	Total
ΔV required, mps	200	2200	100	2500
Min ΔV required, mps	1 ± 0.07			
Firing required per operation	4	1	4	9
Engine				
Propellant	N_2O_4/A-50			
Engine type	Fully ablative or with radiation nozzle (AJ 10-131)			
Engine number	Four-two engines performing MC and OA: all engines performing retro			
Thrust per engine	2200 lb			
Mixture ratio	1.6:1			
Expansion ratio	60:1			
Specific impulse	309 sec			
Combustion pressure	100 psia			
Feed System				
1. Common tankage system Pressurant: helium stored at 4500 psia				
Propellants settled by nitrogen settling jets with nitrogen stored at 3500 psia				

TABLE I-1. VOYAGER GENERAL PROPULSION SYSTEM DESIGN GROUND RULES (Continued)

2. General operating parameters

   Max. propellant temperature: 80° F
   Min. propellant temperature: 40° F
   Nominal propellant tank operating pressure 200 psia
   - Outage 3%
   - Ullage 5%

   Helium and nitrogen pressurants include a 25% leakage and reserve factor

3. Tanks

   All tanks are pressurized in the vicinity of personnel. A safety factor of 1.25 on ultimate loads; in addition, JPL's hazard factor of 1.76 will be used in the design of all pressure vessels, for a total safety factor of 2.2.

   Ti6Al-4V will be used for all helium nitrogen and fuel tanks in the annealed condition.

   For the oxidizer tanks aluminum alloys 2014, 2219 or 6061, Maraging steel or Inconel 718 will be selected. Stress corrosion resistance will govern.

NOTE: Present Ti-6Al-4V LEM Propellant Tanks have an ultimate safety factor of 1.5 in the heat-treated condition. Present VOYAGER working stress will be reduced by a factor of

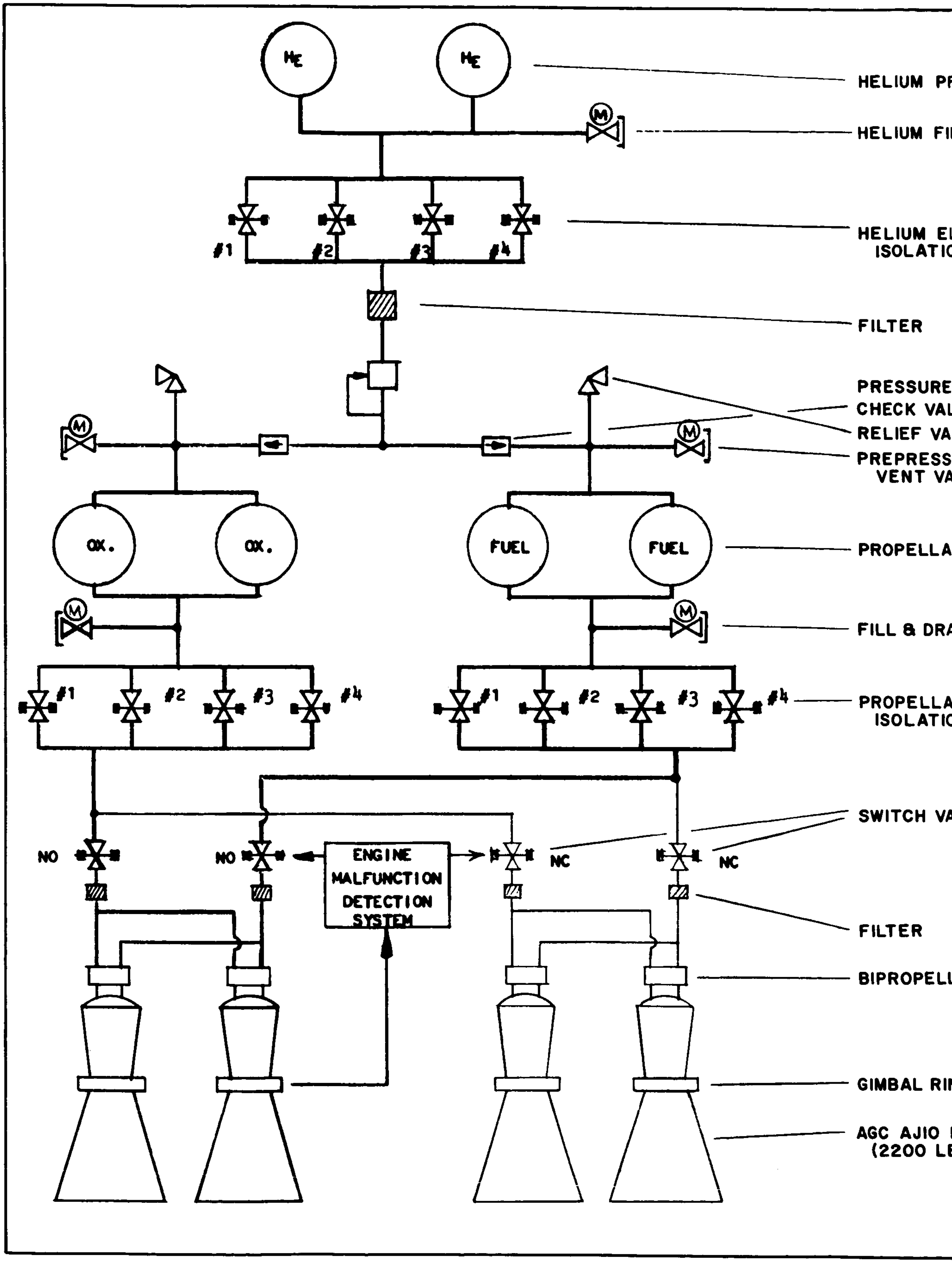
$$\left[\left(\frac{2.2}{1.5}\right) \left(\frac{160,000}{130,000}\right)\right] = 1.8$$

TABLE I-2. VOYAGER PROPULSION SYSTEM WEIGHT SUMMARY

Operation	MC	Retro	OA	Total
Propellant weight required, lb	1,113	8,421	256	9,790
Propulsion system mass-fraction	-	-	-	0.82
Propulsion system weight	-	-	-	11,939
Total impulse, 10^6 lb-sec	0.344	2.602	0.79	3.025

B. SYSTEM DESCRIPTION

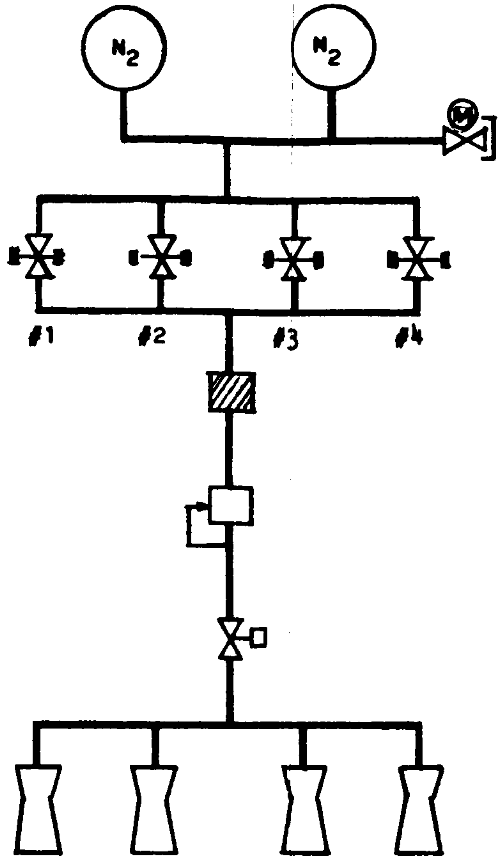
This system, shown schematically in Figure I-1 and illustrated structurally in Figure I-2, uses four AJ10-131 bipropellant engines. A pair of engines performs all MC and OA operations. All four engines are used to perform the retro maneuver. In case of a malfunction, the malfunctioning engine and the one opposite would be shut down, and the remaining engine pair would continue to fire to impart the required velocity increment to the spacecraft. The propellants are N_2O_4 and AeroZINE 50 at a nominal mixture ratio of 1.8 and a combustion pressure of 100 psia. Each engine is fully gimbaled.



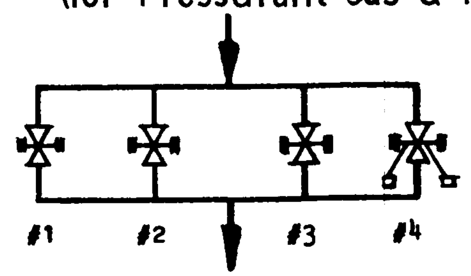
HE
HE
HELIUM PI
HELIUM FI
#1
#2
#3
#4
HELIUM EI
ISOLATIO
FILTER
PRESSURE
CHECK VAL
RELIEF VA
PREPRESS
VENT VA
OX.
OX.
FUEL
FUEL
PROPELLA
FILL & DRA
#1
#2
#3
#4
#1
#2
#3
#4
PROPELLA
ISOLATIO
SWITCH VA
NO
NO
ENGINE
MALFUNCTION
DETECTION
SYSTEM
NC
NC
FILTER
BIPROPELL
GIMBAL RI
AGC AJIO
(2200 LE

ESSURE VESSEL
L VALVE
ECTRO-EXPLOSIVE
N VALVE
REGULATOR
VE
VE
RIZATION &
VE
IT TANK
IN VALVE
NT
N VALVE
VE
ANT VALVE
G
I ENGINE
THRUST)

NITROGEN SETTLING JET SYSTEM



TYPICAL ELECTRO-EXPLOSIVE ISOLATION VALVE SCHEME (for Pressurant Gas & Propellant Isolation)



EACH VALVE WILL HAVE DUAL SETS OF SQUIBS FOR MULTI-CYCLE OPERATION EVEN THOUGH ONLY ONE ON-OFF CYCLE WILL BE USED

VALVE	MISSION	EST. TIME AFTER LAUNCH
NO. 1	1ST MC	2-5 DAYS
NO. 2	2ND MC	30 DAYS
NO. 3	3RD MC & RETRO	200-202 DAYS
NO. 4	FOUR OA	202-247 DAYS

Figure I-1. Voyager Four-Engine Bipropellant System Performing Midcourse Corrections, Retrofiring, and Orbit Adjusts (Common Tankage)

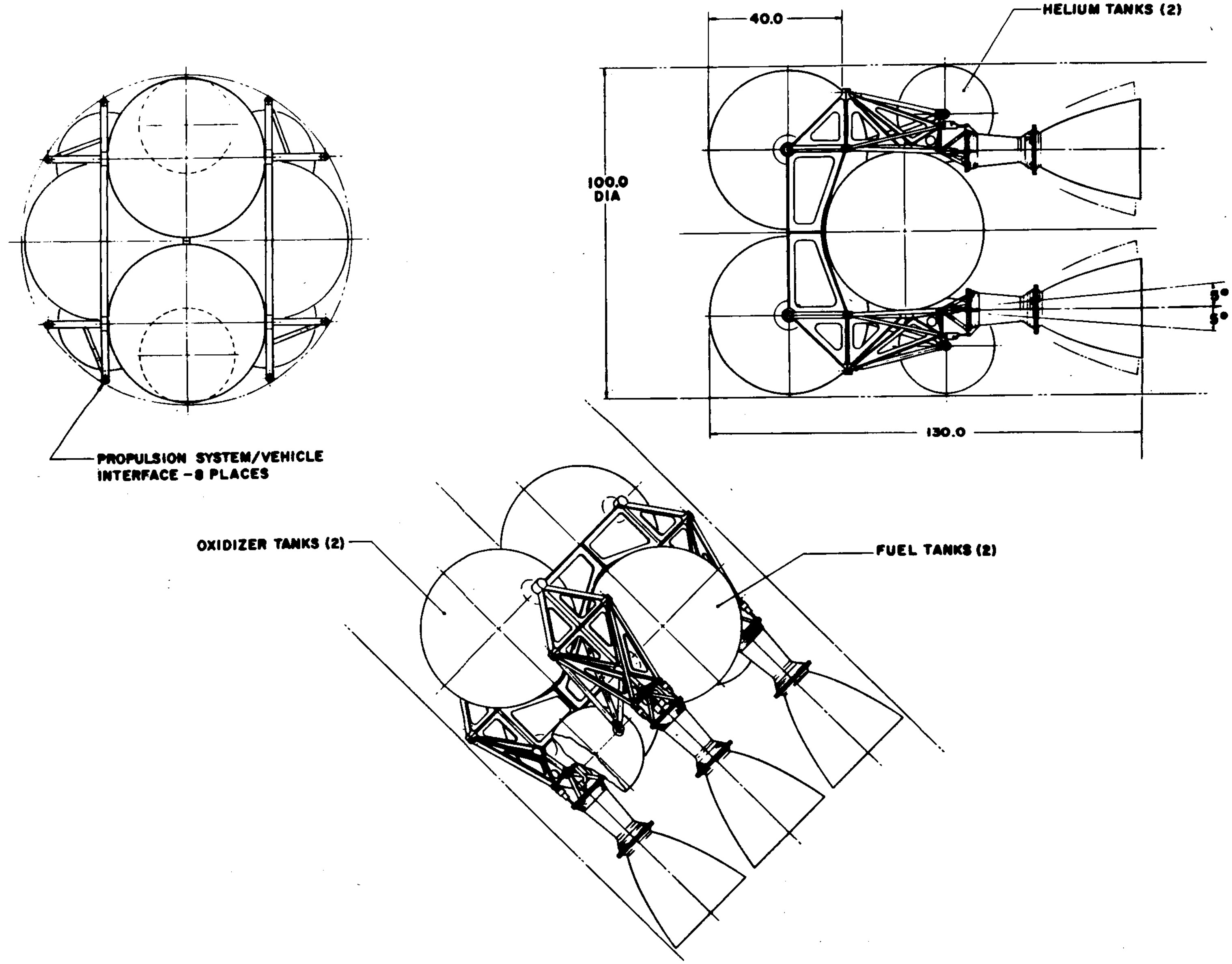


Figure I-2. Voyager Propulsion System-Four-Engine, Bipropellant

The propellants are pressure-fed, and are stored in four equal-size spherical tanks: two for the oxidizer and two for the fuel. Expulsion is in parallel. The propellant tanks will have annular antislosh baffles and antivortex outlet baffles. The pressurant is helium stored in two spherical pressure vessels at 4500 psia and fully regulated to a propellant tank pressure of 200 psia.

The propellants are settled by four nitrogen settling jets firing a nominal duration of 30 seconds prior to each start. Nominal thrust of each nozzle is 12.5 pounds. The nitrogen is stored in two spherical pressure vessels at 3500 psia, and is fully regulated to operating pressure. A solenoid valve controls the flow for each maneuver.

1. Isolation

The pressurant and the propellant are isolated during the long coast periods to minimize leakage. Since it is impractical to provide an isolation valve for each of the possible nine starts, the mission has been divided into four time zones.

- First midcourse correction at launch + 2 days
- Second midcourse correction at +30 days
- Third midcourse correction and retrofiring at +200 and +202 days
- Four orbit adjustments between +202 and +247 days

For each of the four lumped firings an electro-explosive isolation valve has been provided immediately downstream of the helium pressurant, nitrogen settling jet gas and the propellant tank. The valves are normally closed. The electro-explosive valve is opened by firing one squib and closed by firing another squib. Each valve will have a dual set of squibs for multicycle operation even though only one on-off cycle will be used in normal operation. The bipropellant valve, of course, controls the flow for each maneuver, and the pressure regulator does the same for the pressurant.

2. Operation

Loading of propellants and pressurants will be accomplished through manual zero-leakage fill-and-drain valves. Propellant tanks will be prepressurized to 50% of operating pressure through manual valves which also act as vent valves. The pressure regulator has an integral relief valve. The oxidizer and fuel are separated by check valves on the upstream side. Propellant tanks have relief valves set to open below proof pressure. All manual valves are capped, and all relief valves are capped with burst diaphragms. Filters are placed on the upstream side of the pressure regulators and the bipropellant valves.

An engine malfunction-detection system based on combustion chamber pressure, engine temperature, and gimbaling position can shut off a pair of engines during midcourse and orbit adjustment and switch to the other pair by means of electro-explosive switch valves. This process is reversible if the malfunction happened to be corrected.

Prior to each maneuver the nitrogen settling jet system will settle the propellants.

3. Structural Configuration

As shown previously in Figure I-2, the four propellant tanks are pin-mounted in pairs. Each pair is mounted to two beam assemblies by means of diametrically opposite bosses. The four engines are head-gimbaled.

C. COMPONENT AND SUBSYSTEM DESCRIPTION

1. Engine Assembly

A feature of the recommended VOYAGER propulsion system is the use of the 2200-lb-thrust engine which was developed and tested by Aerojet-General during Program 706 (SAINT), the Apollo Service Module Subscale Test Program. The face of this injector is flat, consisting of four concentric annuli, with the oxidizer and fuel being fed through alternate annuli, beginning with fuel on the extreme outside. The annuli are fed from the manifolds through radial feed holes. Performance in excess of 97% of theoretical c* has been obtained consistently with this injector during the Apollo Subscale Test Program.

The ablative chamber is constructed of edge-grain Refrasil fringe tape impregnated with a modified phenyl selaine (Acrylonitrile rubber additive) phenolic resin. The tape is wrapped on a mandrel with an orientation of 60° to the internal contour. The fiber ends at the mandrel surface are directed toward the chamber exit. A phenolic impregnated asbestos felt 0.30-in. thick is used to insulate the liner along its entire length. Aluminum flanges are used for both chamber interface surfaces. The flanges are bonded to a 0.040-in. structural overwrap of glass and glass roving. This overwrap is added to the chamber for longitudinal and hoop strength. Glass overwrap is placed on the flanges for additional strength. Both the glass cloth and roving use epoxy resin as a binder.

A contraction ratio of 3 and a characteristic length (L*) of 30 inches were used in the design of the conical chamber section. The initial expansion region follows the path of an optimum bell contour to match the 60:1 expansion ratio nozzle extension. The ablative portion of the thrust chamber terminates at an exit-to-thrust area ratio of 6.

The all-columbium nozzle extension, is attached to the ablative chamber, and continues the expansion contour to an exit area ratio of 60. The extension is fabricated from columbium alloy C-103, (10% hafnium, 1% titanium), and is 0.030-in. thick for its entire length. The nozzle exterior is diffusion-coated with an aluminide coating to prevent oxygen embrittlement. Design of this extension has drawn heavily upon technology developed and proven in the 624A Transtage and Apollo Service Propulsion System Programs.

Nozzle extension steady-state temperatures during engine firing were calculated utilizing the Bartz equation to determine the heat-transfer coefficient from the exhaust gases to the nozzle wall. The use of the theoretical Bartz heat transfer coefficient has yielded high predicted temperatures for the Apollo Subscale Engine (i.e., predicted temperatures were higher than test results). Therefore, its use for the preliminary calculations of the nozzle temperatures on Voyager should be acceptable since the obtained results represent conservative values.

The forward flange which mates with the combustion chamber attachment flange is fabricated from columbium alloy C-103. The flange configuration is a modified - J which is spin-formed. A metallic seal is utilized on the flange mating surfaces to ensure integrity during high-temperature firing conditions.

2. Thrust Vector Control and Thrust Mount

Conventional engine gimbaling is the selected approach for thrust vector control. The design approach is to use a sealed monoball located at the forward end of the engine assembly. Approximate location is shown by Figure I-2. Flexural pivots will be considered in place of the monoball if control system requirements permit small enough gimbal angle.

3. Valves and Controls

The components selected for the VOYAGER application have been or are similar to those used and qualified for other space programs. Primary selection criteria included adaptability to the VOYAGER mission and duty cycle, proven reliability of design concept, prior use with selected operating fluids, and experience and proven reliability of the selected component vendor.

a. Thrust Chamber Valves

The thrust chamber valve, consists of two identical poppet-type valves, one for fuel and one for oxidizer, integrated in one body and simultaneously operated by means of a hydraulic actuator and conventional stabilizing linkage. This bipropellant valve arrangement is that used successfully on the SAINT Apollo Subscale and VOYAGER Research Engine. It will be used for the VOYAGER application with any modification determined necessary for compatibility with the space environment and mission duty cycle.

The valve opening cycle consists of energizing a three-way solenoid pilot valve which directs fuel system pressure to the actuator. When the actuator is vented by de-energizing the solenoid pilot valve, the system line pressure and valve spring forces will cause the valves to close.

b. Engine Isolation Valves

The pressurant and the propellant are separated from their respective discharge manifolding by tank isolation valves during long vehicle coast periods. The isolation valve design uses squib-operated devices using the design principles of the squib valves produced by SieBelAir.

c. Pressure Regulator

The pressure regulator is a scaled-up version of a regulator qualified for use on the Transtage ACS, Gemini RCS and OAMS system, and employs the same design principles as one to be qualified on the lunar orbiter program. Modification of the regulator consists of enlarging the throttling orifice for greater flow, decreasing the output pressure slightly, incorporating a regulated pressure relief function to prevent over pressurization of the propellant tanks in the event of a regulator malfunction of an existing pressure surge protector.

d. Relief Valves

The relief valve is the same design as a valve qualified for use on the Surveyor program, and presently being qualified for use on the Apollo program at Aerojet-General. The design has been modified to change the main pressure sensing spring to facilitate operation at a different pressure, replace the dynamic piston seal with a spring-loaded Teflon seal, and provide coating of the dynamic piston to prevent cold-welding of sliding surfaces. The dynamic piston seal presently consists of a silicon O-ring compressed behind a Teflon ring which slides against the piston to provide a low friction seal for minimum hysteresis.

e. Filters

The filters employed throughout the system are rated at 2 micron nominal and 15 micron absolute. The pressurization system filter is incorporated in the pressure regulator inlet. The propellant system filters will be inline welded units. All of the filters will be of the electroetched stacked-washer type. This type of filter consists of a stack of segments resembling thin washers, each of which has one face chemically etched to provide a predetermined intricate flow path. The stack of segments is held rigidly and is tightly compressed.

f. Manual Fill-and-Drain Valves

Manually operated valves are used in all system locations and is designed to be welded into the system. The design is constructed of all-stainless nonmagnetic steel with a Teflon seat swaged into the inlet fitting, and is capable of operation up to 6000 psi.

4. Propellant Tanks

The tank system proposed is shown schematically by Figure II-1, and general arrangement is shown in Figure II-2.

The four propellant tanks are pin-mounted in pairs. Each pair is mounted to two beam assemblies by means of diametrically opposed bosses. The axes of the tank pairs are orientated 90° to each other so that the four beams form an open cross intersecting at four points. All tank loads and engine thrust loads are transmitted to the vehicle through the ends of each beam. These eight points lie in a common plane perpendicular to the axis of the vehicle, and form the propulsion system/vehicle structural interface. The four gas bottles (two helium and two nitrogen) are supported by their two bosses with a tubular structure which transmits their acceleration loads to the tank mounting beams. The engine is supported by the two thrust pads on the gimbal assembly by means of four tubular struts on each side. These struts transmit the engine thrust loads to the two oxidizer tank support beams.

Propellant tanks will be spherical and contain the necessary bosses for pressurization, propellant discharge, and mounting. Tank design will be based on the technology developed by Aerojet in the manufacture of tanks for the Apollo LEM Ascent stage.

All tanks except the oxidizer tank will be fabricated from Ti-6A1-4V in the annealed condition, using an ultimate safety factor of 1.25 times JPL's hazard factor of 1.76. Because of the present uncertainty of using Ti-6A1-4V with N_2O_4, other materials will be investigated such as aluminum alloy 2014-T6 and 6061-T6, Inconel 718, and 18% maraging steel.

Propellant tanks will include aluminum anti-slosh baffling.

5.0 GUIDANCE AND CONTROL. This section discusses the effects of the various propulsion systems and resulting configurations on the vehicle attitude control and autopilot. Differences in the autopilot designs for the alternatives considered are not significant. In the case of attitude control, however, the large mass of liquids associated with the transtage or Lunar Excursion Module Descent Equipment (LEMDE) are of major concern.

5.1 Comparison of Attitude Control Subsystem Configurations. The functions of the preferred design for the Attitude Control Subsystem (ACS) are defined in VC234FD105, Volume A and are summarized as follows:

a. Initially stabilize to a Sun-Canopus celestial reference system.
b. Maintain this attitude during transit and areocentric orbit.
c. Slew the Spacecraft to various inertial attitudes as required for trajectory corrections.
d. Maintain roll control during the orbit insertion maneuver.
e. Reacquire celestial references following autopilot operation.
f. Slew to a desired inertial attitude and hold during Capsule separation.
g. Maintain inertial attitude during occultation of celestial references.

The ACS functions in the transtage and LEMDE designs would be identical to these except for item (d). In the case of transtage, no roll control is required of the ACS during the orbit insertion maneuver, while in the LEMDE design the ACS must provide roll control during all trajectory corrections.

Since the ACS performs essentially the same functions as in the preferred design, the same basic system would be used. The moments of inertia in the LEMDE design are slightly higher than those of the preferred design, while in the transtage design they are 1.7 to 5.0 times greater, depending on the phase of the mission. Thus, the amount of cold gas required for limit cycle operation will be much greater in the transtage design.

The transtage and LEMDE ACS performance may differ significantly from that of the preferred design because of the relatively large amount of liquid propellants used in these configurations. In the preferred design, the nonrigid mass is only a small fraction of the total mass so that rigid body analysis appears justified. Such is not the case in the transtage and LEMDE designs, however. In order to accurately predict the ACS performance, an experimentally and/or analytically determined mathematical model describing the propellant motion is required. This model could not be developed in the time period of this study.

Since the vehicle transfer function in the preferred design will be well defined with minimal analysis and test, control system parameters which provide adequate performance during all mission phases may be determined with a high degree of confidence. This is the major advantage of the preferred design over the transtage and LEMDE designs from an ACS standpoint, and was a major reason for the choice of the solid rocket.

5.2 Comparison of Autopilot Configurations.

5.2.1 General Description. This section makes a comparative evaluation of the autopilot for each of the following configurations:

a. Preferred design (Solid Retro)
b. Design based on transtage engines
c. Design based on LEMDE

The preferred design autopilot has been described in VC234FD105, Volume A, while the other two designs are described herein.

5.2.1.1 Transtage. As shown in Figure 5.2-1, the transtage design incorporates four gimballed bipropellant engines for low thrust maneuvering in addition to the two transtage engines. Orbit insertion, as well as large correction maneuvers, is accomplished by the large engines. It is shown in Section 5.2.3, that the velocity error for small velocity corrections becomes excessive with the transtage engines, necessitating the low thrust engines.

It is not planned to operate both propulsion systems at the same time. However, the autopilot must be capable of controlling the vehicle when only two diametrically opposed low thrust engines are operating. This situation can occur if an engine fails and the diametrically opposed one is shut down. Such an operation is identical to the midcourse system of the preferred design.

To promote simplicity and reliability, the autopilot has been configured to minimize switching and other internal parameter variations, such as gain changes while maintaining an acceptable response. For example, only limited switching operation in the autopilot is required to accommodate the propulsion system being operated. Separate amplifiers with appropriate gains and compensation circuits are provided for the actuators for both types of propulsion system.

When the main transtage engines are operated, the autopilot commands the small engine actuators as well as the large engine gimbal actuators. Of course, since the small engines will not be operating at this time, the operation of their gimbals will have no control effect. The inertial coupling due to motion of the small engines has been demonstrated to be negligible. This approach is felt to be more reliable than providing special switching to turn off the signal to the small engine actuators during orbit insertion.

During small correction maneuvers when the small engines are operational, however, it will be necessary to provide a switching function so that the transtage gimbal actuator will not be operated. This represents a departure from the approach of the preferred design and is necessary, to:

a. Enhance reliability of the gimbal actuator
b. Avoid the relatively high power requirements of the large gimbal actuator
c. Avoid the engine inertial coupling (i.e., "Tail-wags-dog") effect of the large engine during operation of the smaller engines.

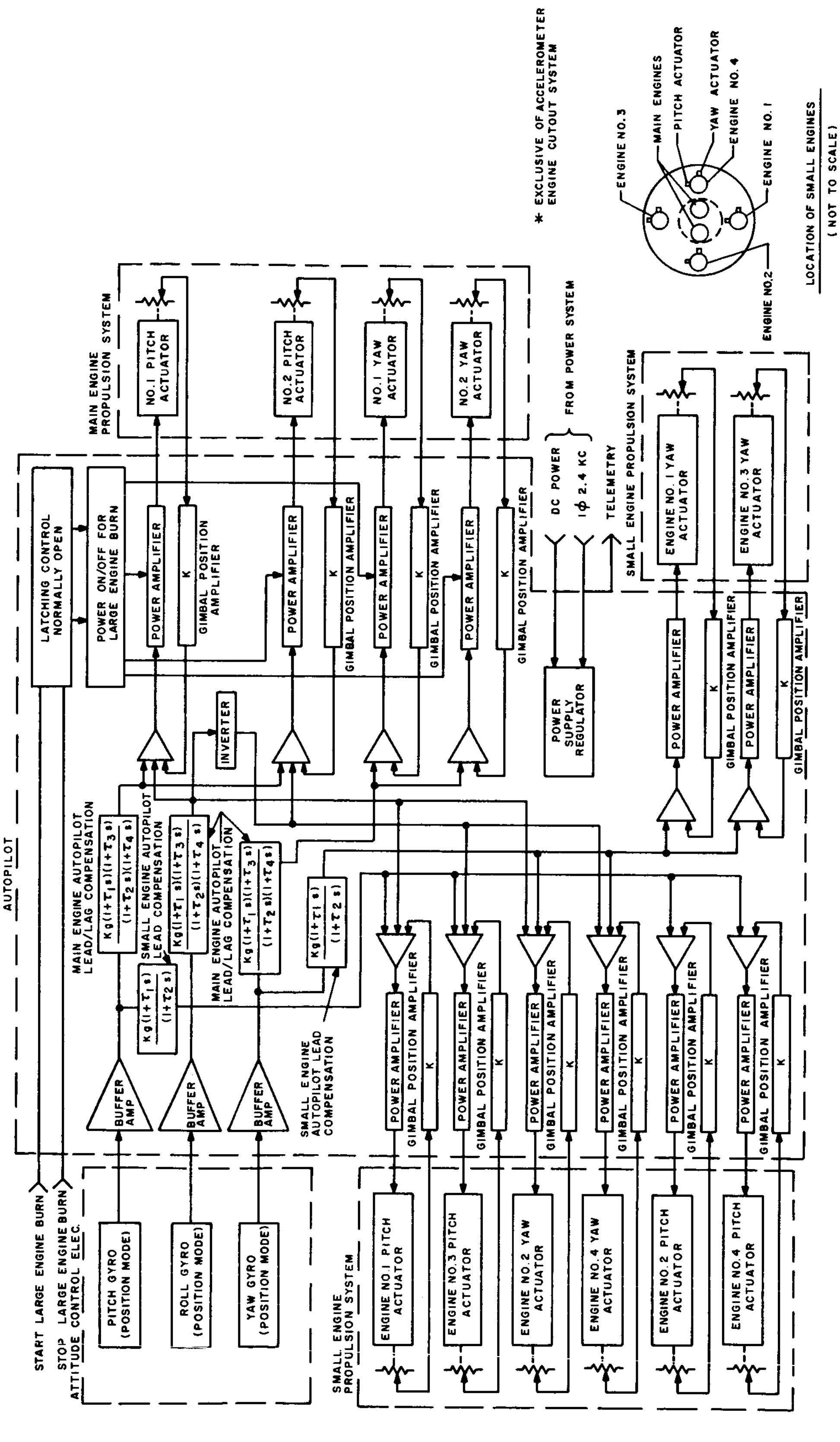


Figure 5.2-1. Transtage Block Diagram

As discussed in the propulsion section, pitch and yaw control torque is derived during retro-fire by actuation of the pitch and yaw gimbals of the two main engines. Roll control torque is obtained by differential operation of the pitch gimbals. The geometry is shown in Figure 5.2-2 with the engine displacement sign conventions indicated for positive pitch, yaw, and roll displacements.

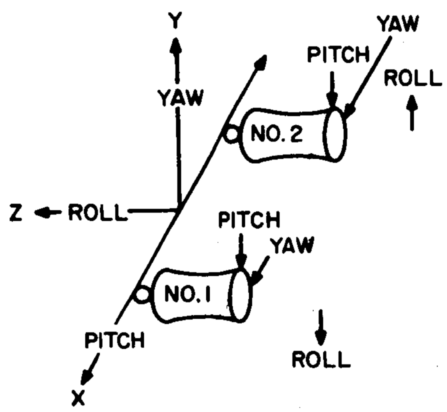


Figure 5.2-2. Transtage Dual Engine Geometry

The gimbal actuators used by the present transtage are high torque hydraulic actuators and have not been demonstrated for applications involving long exposure to hard space environment. A redesign of the actuators to all electric torquers may be required, unless hydraulic actuators operating at high torque levels can be demonstrated to be storable in a hard space environment. Storage periods equal to the total transit lifetime must be satisfied.

Control torque generation during low thrust engine operation is similar to that described in VC234FD105, Volume A for the preferred design. Four bipropellant engines, each with two separate gimbals, provide redundant pitch, yaw, and roll control for the correction maneuvers. These engines use the same propellants as the main engine. Each gimbal has a separate amplifier and position feedback. The autopilot provides for differential gimbal operation to provide roll control.

The magnitude of all velocity increments with the transtage design would be controlled by integrating the output of a roll-axis accelerometer and comparing it to a preset stored value. When the desired velocity is attained, a stop-engines command is issued to the propulsion subsystem. This technique is used in the preferred design for midcourse and orbit adjust maneuvers.

5.2.1.2 LEM Descent Engine (LEMDE). Figure 5.2-3 is a functional diagram of the LEMDE autopilot. The basic LEMDE is a continuously throttlable gimballed bipropellant engine. The continuous throttling capability is not required for VOYAGER. As shown in Section 5.2.3, the velocity increment accuracy can be met with a two-thrust level system. Orbit insertion and large correction maneuvers are performed at the maximum thrust level (10,500 pounds); while for small correction maneuvers, the engine is throttled to its minimum thrust level (1050 pounds). This discrete throttling can be accomplished more reliably than continuous throttling.

To promote simplicity and reliability, the autopilot has been configured so as to minimize switching and other internal parameter variations (e.g., gain changes) and still have an acceptable response. For example, only limited switching operation in the autopilot is required to accommodate the thrust level being operated. Separate amplifiers with appropriate gains and compensation circuits are provided for both thrust levels.

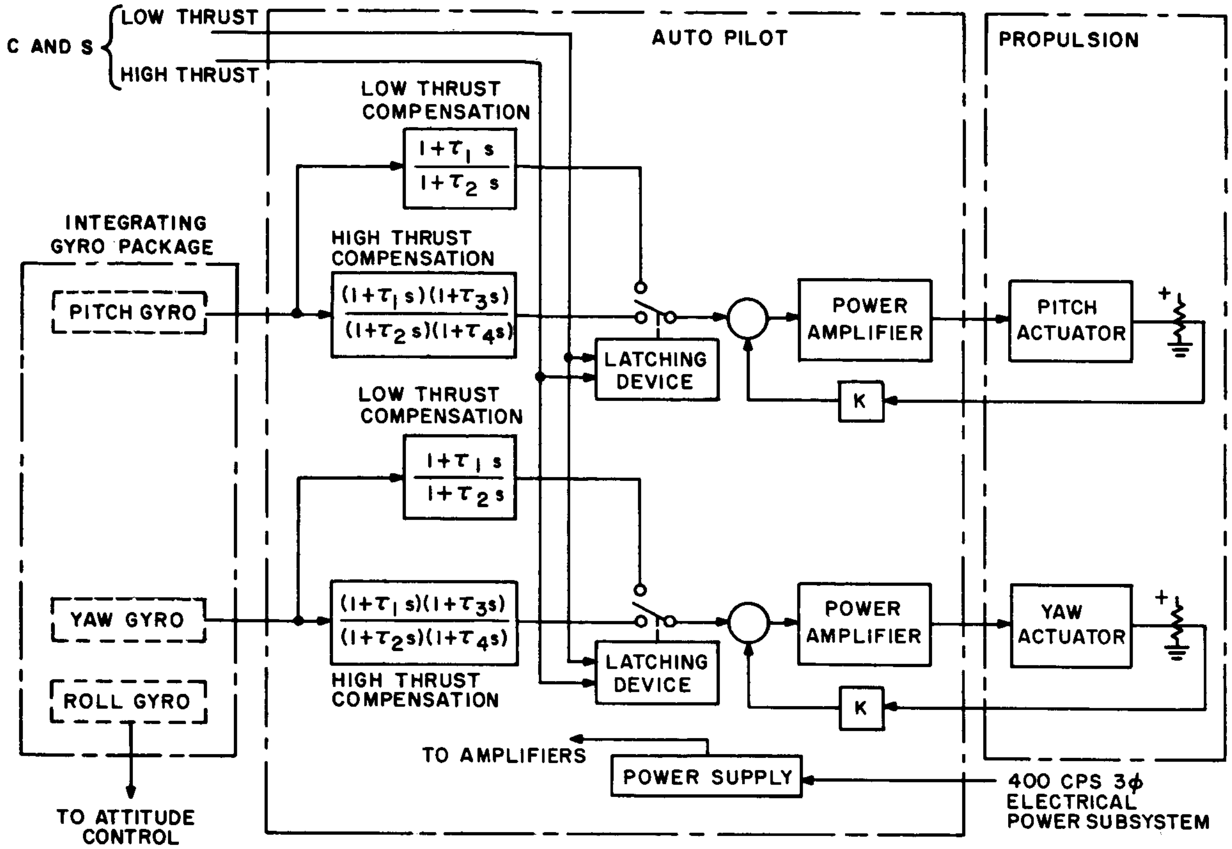


Figure 5.2-3. LEMDE Functional Diagram

As discussed in the propulsion section, pitch and yaw control torque is derived by actuation of the pitch and yaw gimbals of the liquid engine. Roll control torque is supplied by the ACS during all maneuvers. Control of the magnitude of all velocity increments is identical to that of transtage and the small engines of the preferred design.

5.2.2 Comparison Summary. Table 5.2-1 summarizes the techniques used for obtaining control torques in each of the three basic configurations.

TABLE 5.2-1. CONFIGURATIONS COMPARED ON BASIS OF THEIR CONTROL TORQUING MEANS

Design	Correction Modes		Orbit Insertion Mode	
	Pitch/Yaw Torques	Roll Torques	Pitch/Yaw Torques	Roll Torques
Preferred	Monopropellant engine vanes	Monopropellant engine vanes	Secondary injection	ACS roll jets
Transtage	4 gimballed engines	Differential operation of gimbals of small engines	2 gimballed engines	Differential operation of gimbals of main engine
LEM Descent Engine	Gimballed main engine (throttled to low thrust)	Attitude control system roll jets	Gimballed main engine (at high thrust)	Attitude control system roll jets

The Minuteman secondary-injection actuator appears acceptable for VOYAGER without modification. In the case of the other designs, however, a redesign of the gimbal actuators is anticipated, for reasons of long-term reliability in the case of transtage and to improve response time in the case of LEMDE. Thus, from an implementation standpoint, the preferred design has the advantage of tested mechanizations.

From an autopilot standpoint, there are three basic factors which cause the transtage and LEMDE designs to differ from the preferred design, in addition to the difference in means of generating control torques:

a. Spacecraft Configuration - The moments of inertia and location of the center of mass and gimbal point

b. Propellant Motion

c. Engine Inertial Coupling

The Spacecraft characteristics are discussed in Appendix A and are summarized in Table A-1. Referring to this table, it can be seen that the vehicle gain (thrust times moment arm divided by inertia) varies: over a 4.4 to 1 range with the LEMDE configuration, over a 4.3 to 1 range with the unmodified transtage configuration, and over a 3.3 to 1 range with the modified transtage configuration for both large and small engine operation. In the case of the preferred design, the vehicle gain varied over a 7.7 to 1 range with the solid engine and over a 10 to 1 range with the small engines. It was demonstrated in Volume A, however, that adequate performance is obtained with passive compensation techniques even with a 20 to 1 gain variation, so that no significant advantage accrues from the smaller gain variation.

It is anticipated that the lateral shift of the Spacecraft center of mass will be greater with the transtage design than with the other designs due to the asymmetry of the transtage tank shapes. For this reason pre-positioning of the thrust vector may be required to minimize the turn-on transient.

In the transtage and LEMDE designs, the mass of the liquid propellants represent a significant portion of the total Spacecraft mass so that the effect of propellant motion will be much greater than in the case of the preferred design. In addition, the most critical case will be the first midcourse correction, since the liquid propellant mass is maximum at that time, whereas in the preferred design, the fuel sloshing effect is most critical in the orbit adjust mode, after the solid fuel has been depleted and the capsule has been ejected. An analysis of the fuel sloshing effect has not been completed, but it is anticipated that in the case of the transtage and LEMDE designs, a more complex compensation network will be required. Subsequent analysis in this report does not consider this effect.

As discussed in the propulsion section, propellant motion can be restricted by judicious placement of screens in the propellant tanks. This technique has not yet been proven, however, and would require a significant test program to demonstrate its feasibility for use on VOYAGER.

The engine inertial coupling effect is discussed in Appendix B, and is included in the analysis of Section 5.2.3. It is shown in Section 5.2.3 that satisfactory autopilot performance is obtained with this effect present so that the inertial coupling effect does not appear to be a serious disadvantage except in the transtage design, where it necessitates a switching function to disable the main engine actuators during operation of the small engines.

5.2.3 Analysis

5.2.3.1 Thrust Pointing Control. Figure 5.2-4 is a single-axis servo diagram of the autopilot. This generic block diagram is applicable to both the transtage and LEMDE designs, and differs from the preferred design servo diagram only by the addition of the engine inertial coupling effect, which is discussed in Appendix B.

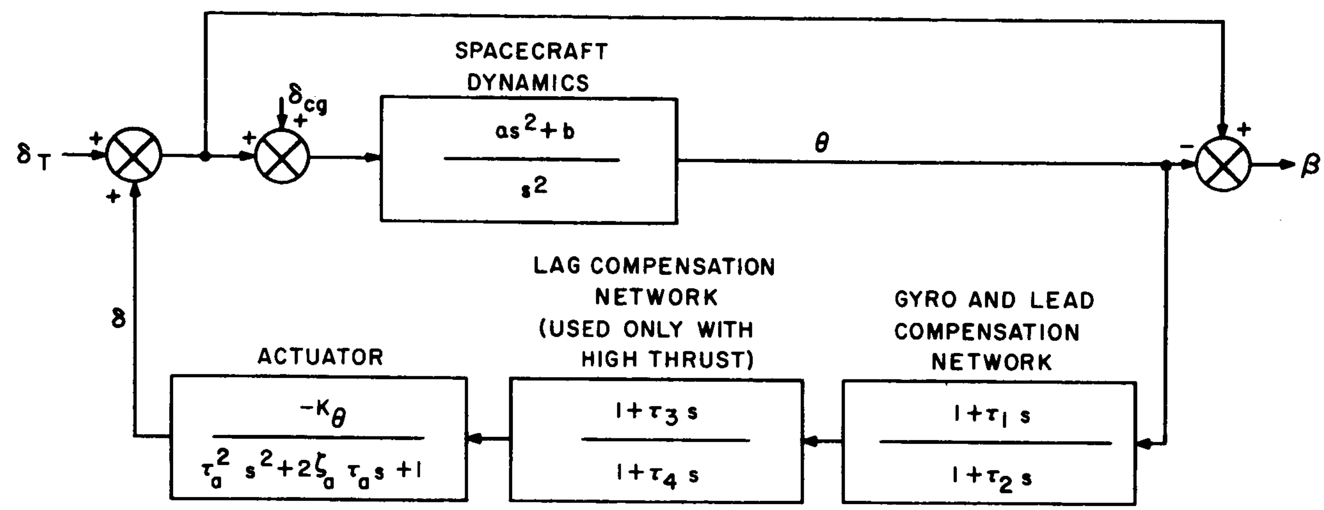


Figure 5.2-4. Autopilot Servo Diagram

The angles are defined as follows:

- β is the thrust pointing error with respect to the inertial reference direction
- θ is the spacecraft attitude with respect to the inertial reference direction
- δ is the angle through which the thrust vector is deflected, measured from its initial position (gimbal angle)
- δ_T is the thrust misalignment angle; i.e., the angle between the initial thrust vector position and the spacecraft roll axis
- δ_{cg} is the uncertainty component of angular offset of the spacecraft center of mass from the nominal thrusting axis. The steady-state error due to the corresponding predictable component is compensated for by appropriately modifying the commanded angular turns.

The feedback gain, K_θ, includes the dc gain of all the elements in the feedback loop; i.e., the gyro, compensation network, and actuator. The vehicle gain, b, as well as the engine inertial coupling coefficient, a, is given in Table A-1 for the various phases of the mission for both the transtage and LEMDE configurations. The lag network is included only in the compensation network for high thrust level operation, to reduce the inherently higher bandwidth.

Figure 5.2-5 defines the sign convention used in the subsequent discussion. All angles are positive as in the indicated directions.

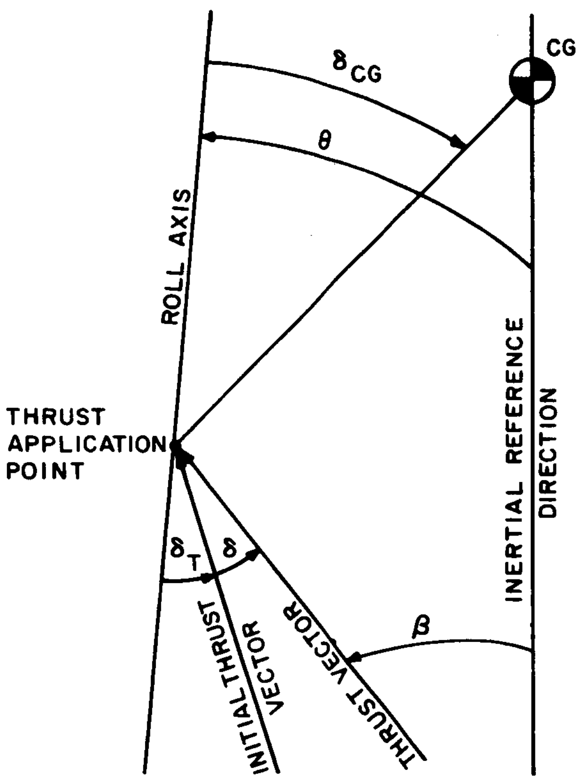


Figure 5.2-5. Sign Convention

Referring to Figure 5.2-4, the open-loop transfer function for the autopilot is given by

$$GH = \frac{(a\,s^2 + b)\,K_\theta\,(1 + \tau_1 s)\,(1 + \tau_3 s)}{s^2\,(1 + \tau_2 s)\,(1 + \tau_4 s)\left[\tau_a^2 s^2 + 2\zeta_a \tau_a s + 1\right]}$$

The thrust direction response is given by:

$$\beta = \delta - \theta + \delta_T \qquad (5.2\text{-}1)$$

It is important to recognize from this equation that thrust direction is directly a function of both vehicle attitude and thrust deflection angles relative to the vehicle. For this reason, a very high gain autopilot may rapidly drive the vehicle attitude error toward zero at the expense of large transient excursions in δ and correspondingly large transient errors in β. Likewise, a very low gain may eliminate large excursion in δ, but at the expense of large excursions in θ. Selection of system gain is particularly critical from the standpoint of minimizing total thrust pointing errors during short correction maneuvers. Steady state response, as discussed below, is also an important consideration.

The steady-state thrust pointing error, β_{ss}, is given by

$$\beta_{ss} = \delta_T + \delta_{ss} - \theta_{ss} \qquad (5.2\text{-}2)$$

In the steady-state, the net torque on the spacecraft must be zero, so that

$$\delta_T + \delta_{ss} = -\delta_{cg} \qquad (5.2\text{-}3)$$

Thus,

$$\beta_{ss} = -\delta_{cg} - \theta_{ss} \qquad (5.2\text{-}4)$$

The spacecraft response to the disturbance torques resulting from the relative center of mass offset and the thrust misalignment is given by

$$\frac{\theta}{\delta_T + \delta_{cg}} = \frac{(as^2 + b)(1 + \tau_2 s)(1 + \tau_4 s)(1 + 2\zeta_a \tau_a s + \tau_a^2 s^2)}{s^2 (1 + \tau_2 s)(1 + \tau_4 s)(1 + 2\zeta_a \tau_a s + \tau_a^2 s^2) + (as^2 + b) K_\theta (1 + \tau_1 s)(1 + \tau_3 s)} \tag{5.2-5}$$

The steady-state spacecraft attitude resulting from a step input disturbance is given by

$$\theta_{ss} = \lim_{s \to o} s\, \theta(s) = \frac{\delta_T + \delta_{cg}}{K_\theta} \tag{5.2-6}$$

Substituting Equation (5.2-6) into Equation (5.2-4)

$$\beta_{ss} = -\delta_{cg} - \left(\frac{\delta_T + \delta_{cg}}{K_\theta}\right) \tag{5.2-7}$$

As discussed in VC234FD105, Volume A, the autopilot pointing accuracy goal is ± 0.5 degree (3σ) on a single-axis basis. Substituting a thrust misalignment of 0.25 degree, and an assumed angular center of mass offset of 0.25 degree into Equation (5.2-7), it is seen that the feedback gain chosen in the preferred design (K_θ = 5.0) is again adequate from a steady-state error standpoint.

Assuming a feedback gain of 5.0, the attenuation-phase diagrams corresponding to the transtage and LEMDE autopilots during the retromaneuver are shown in Figures 5.2-6 and 5.2-7, respectively. As in the preferred design, the passive compensation network provides adequate phase margin throughout the range of vehicle gains. The time constants used in the compensation network are shown in Table 5.2-2.

The transient error during the retromaneuver was evaluated by examining the autopilot response to a step disturbance input. An angular cg offset of 0.25 degree was used. The resultant spacecraft attitude (θ), gimbal angle (δ) and thrust pointing error (β) are presented in Figures 5.2-8 through 5.2-11. Figures 5.2-8 and 5.2-9 correspond to the extreme values of vehicle gain with the transtage design, while Figures 5.2-10 and 5.2-11 are the corresponding curves for the LEMDE design. Comparing these figures to the corresponding figures for the preferred design (Figures 3-14 through 3-16 of VC234FD105, Volume A) reveals that the responses are very similar. As in the case of the preferred design, the initial transient subsides rapidly and is averaged out over the retrothrusting time.

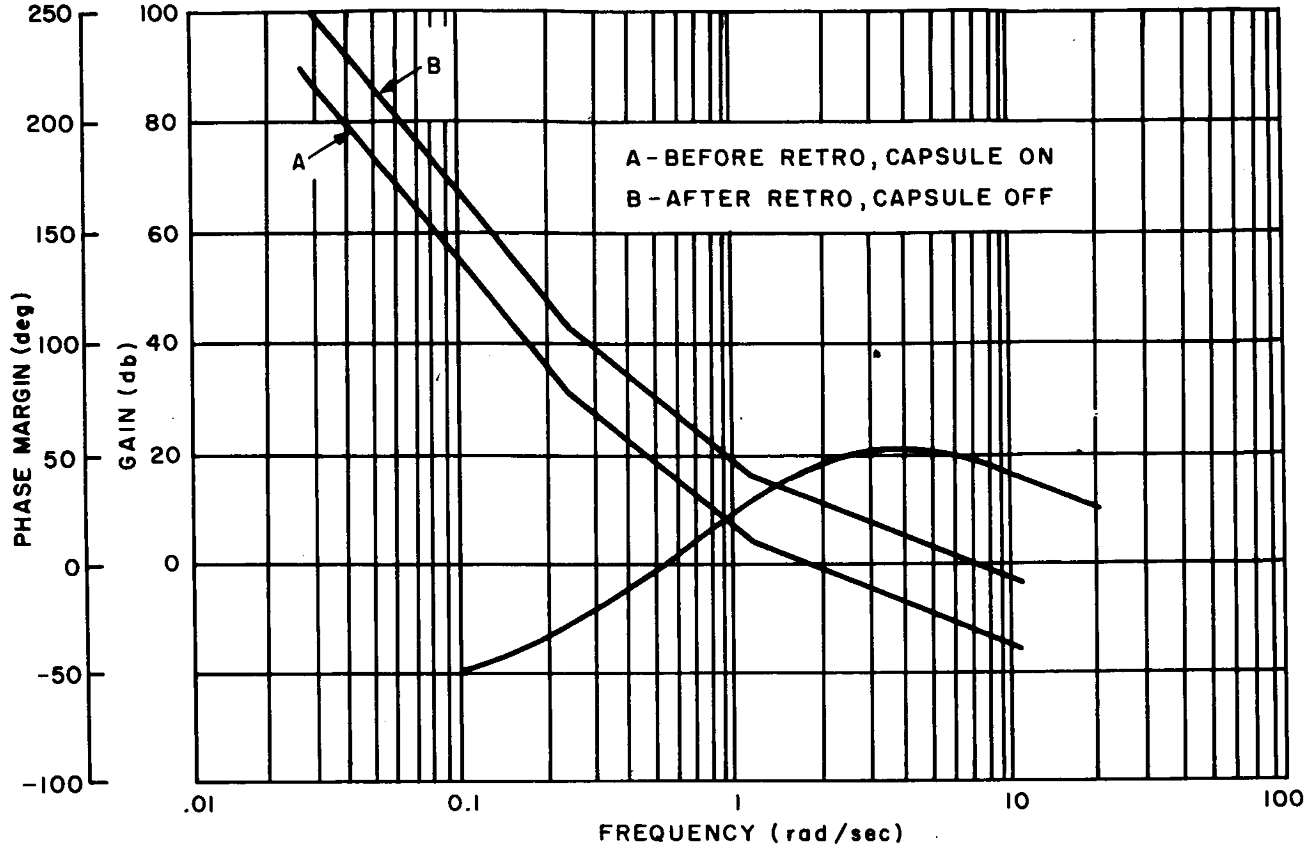


Figure 5.2-6. Transtage Attenuation-Phase Diagram

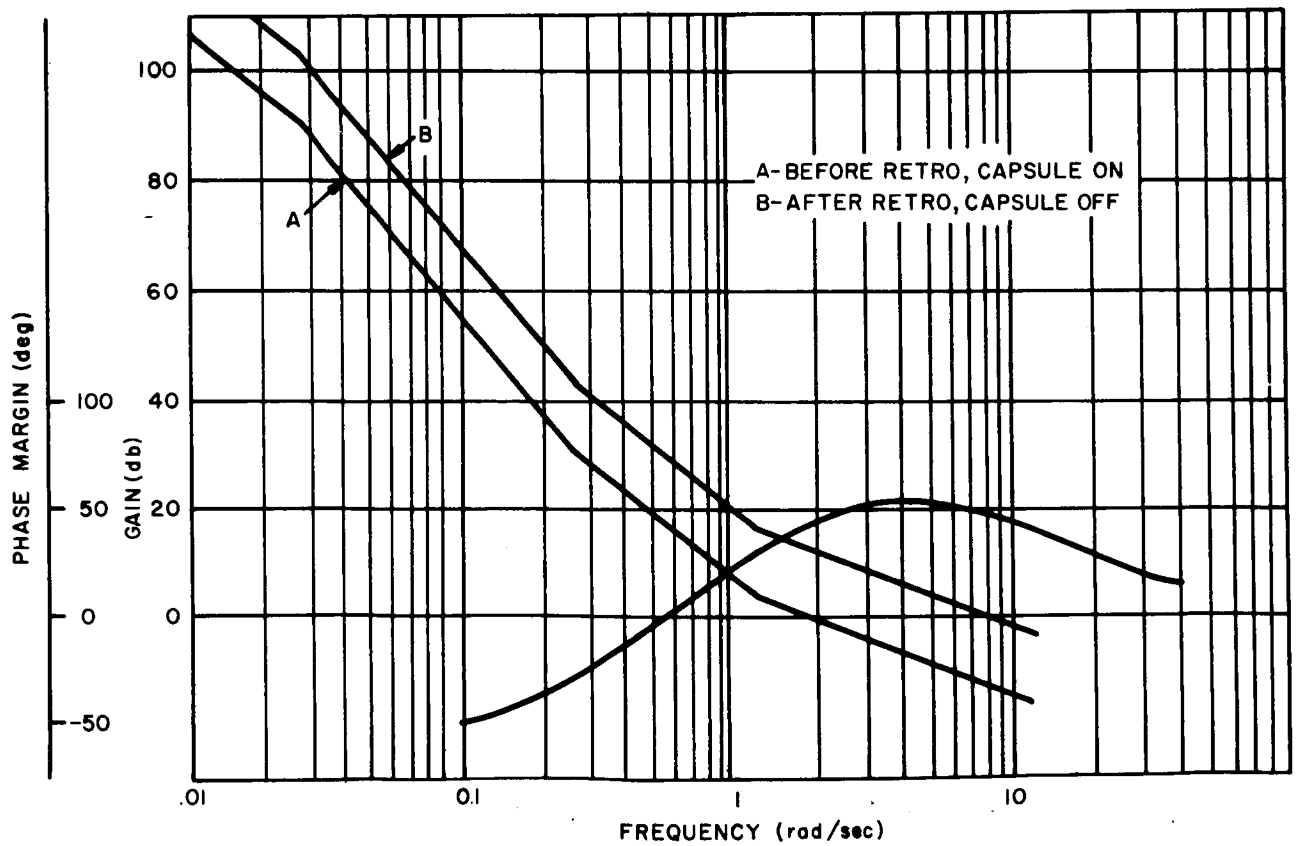


Figure 5.2-7. LEMDE Attenuation-Phase Diagram-High Thrust Mode

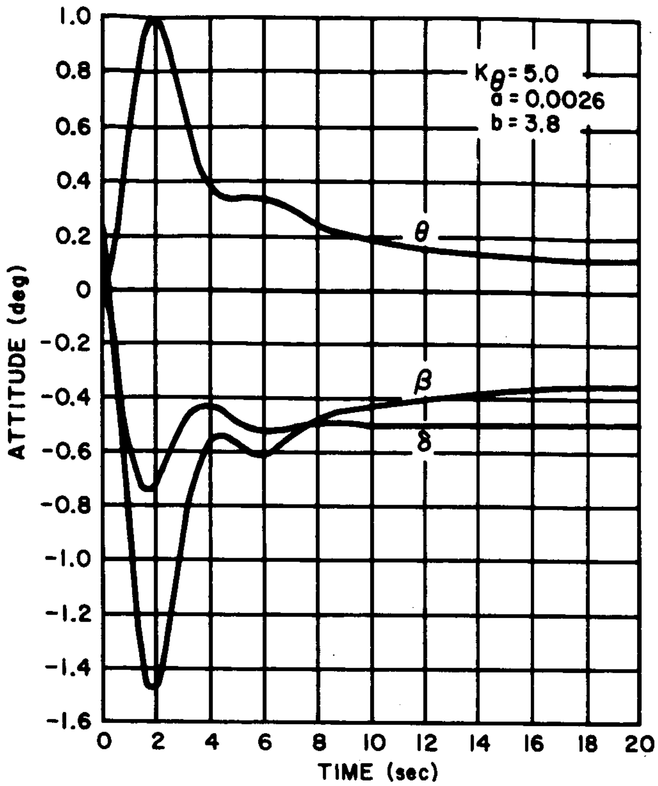


Figure 5.2-8. Transtage Autopilot Response-Start of Retro with Capsule On

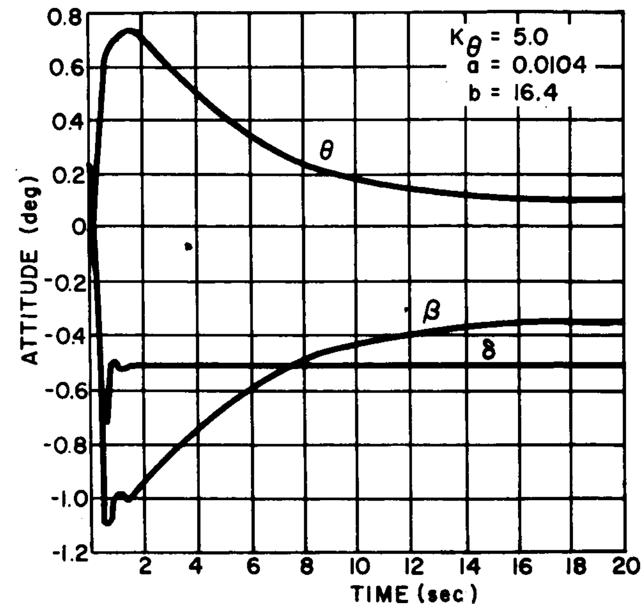


Figure 5.2-9. Transtage Autopilot Response-End of Retro with Capsule Off

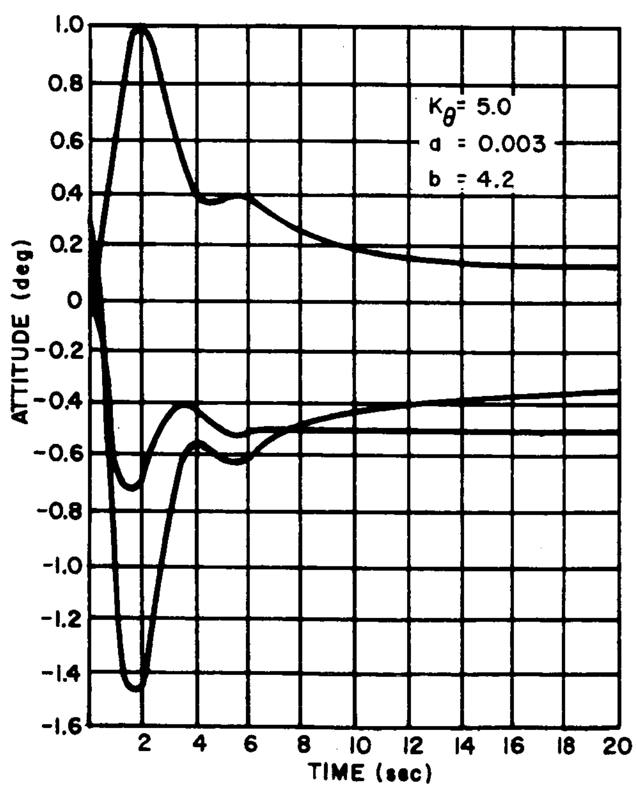


Figure 5.2-10. LEMDE Autopilot Response-Start of Retro with Capsule On

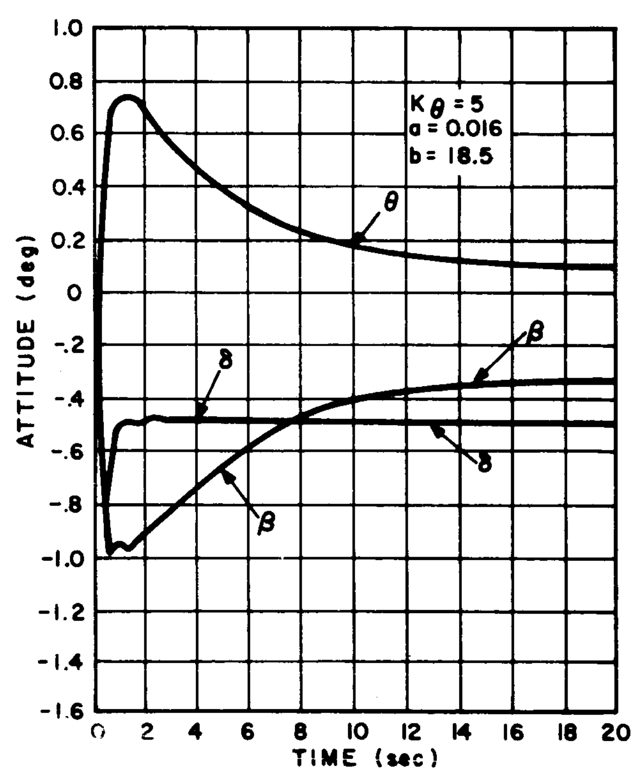


Figure 5.2-11. LEMDE Autopilot Response-End of Retro with Capsule Off

TABLE 5.2-2. TIME CONSTANTS USED IN COMPENSATION NETWORK

Configuration	K_θ	τ_1 (sec)	τ_2 (sec)	τ_3 (sec)	τ_4 (sec)
Transtage	5.0	0.875	0.0875	4.0	40.0
LEMDE	5.0	0.833	0.0833	4.0	40.0

Figures 5.2-12 and 5.2-13 represent the attenuation-phase diagrams corresponding to the transtage and LEMDE autopilots during small engine operation. Again adequate phase margin exists throughout the range of possible vehicle gains. The time constants used in the lead compensation network are given in Table 5.2-3.

Again the transient response was evaluated and is plotted in Figures 5.2-14 through 5.2-17. Referring to these figures, it can be seen that the results are again similar to those obtained with the preferred design (Figures 3-27 through 3-29 of VC234FD105, Volume A). The overshoot is relatively small, so that the steady-state error provides a good measure of the pointing error, even during small velocity corrections.

In summary, the thrust pointing control performance of the transtage and LEMDE autopilots is similar to that of the preferred design if fuel sloshing effects are ignored. Because of the relatively large mass of liquid propellants associated with transtage and LEMDE, however, it is anticipated that the fuel sloshing will necessitate a more complex compensation network.

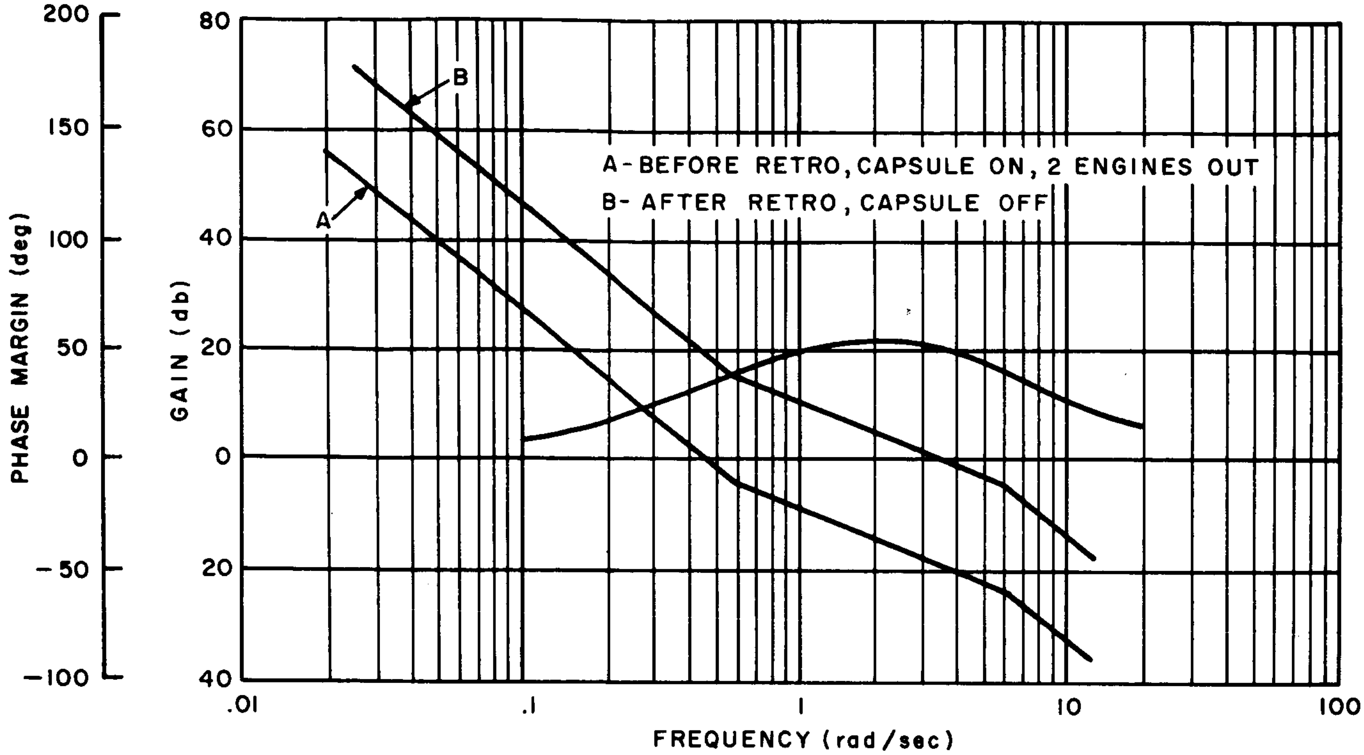


Figure 5.2-12. Transtage Attenuation-Phase Diagram, Small Engines

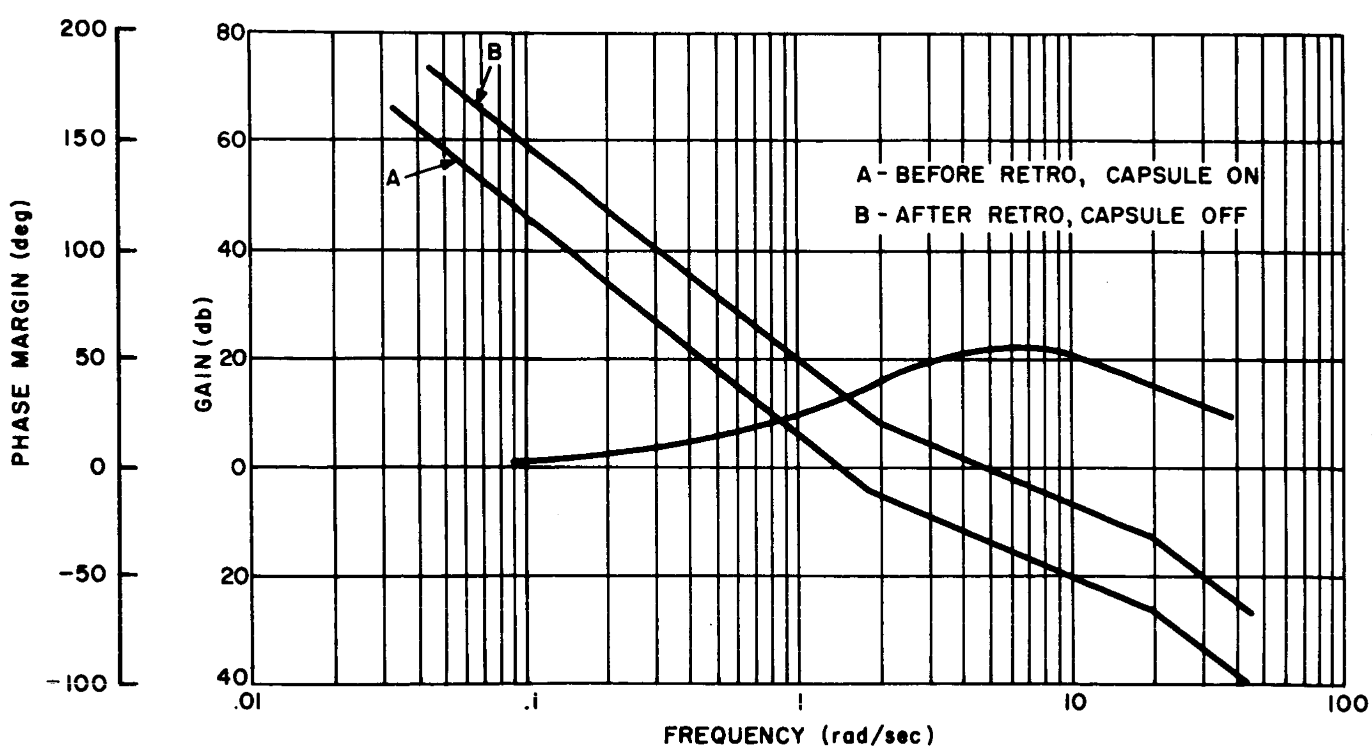


Figure 5.2-13. LEMDE Attenuation-Phase Diagram, Low Thrust Mode

TABLE 5.2-3. TIME CONSTANTS FOR LEAD COMPENSATION NETWORK

Configuration	K_θ	τ_1	τ_2
Transtage	5.0	1.65	0.165
LEMDE	5.0	0.5	0.05

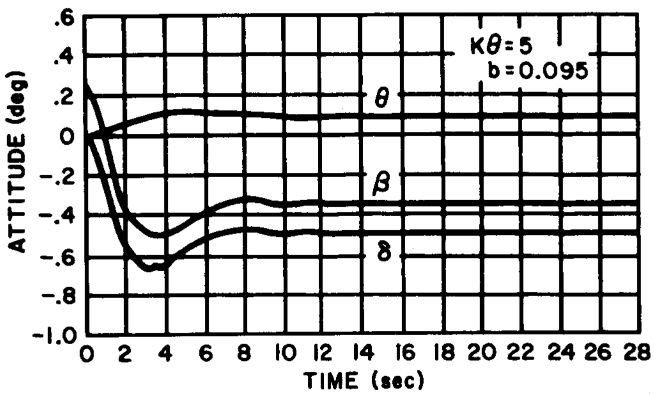


Figure 5.2-14. Transtage Autopilot Response-Prior to Retro with Capsule On

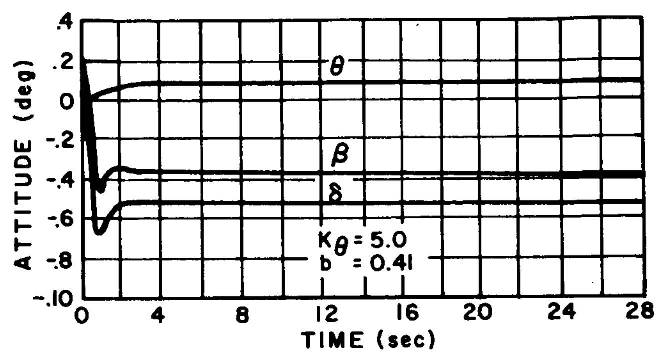


Figure 5.2-15. Transtage Autopilot Response-After Retro with Capsule Off

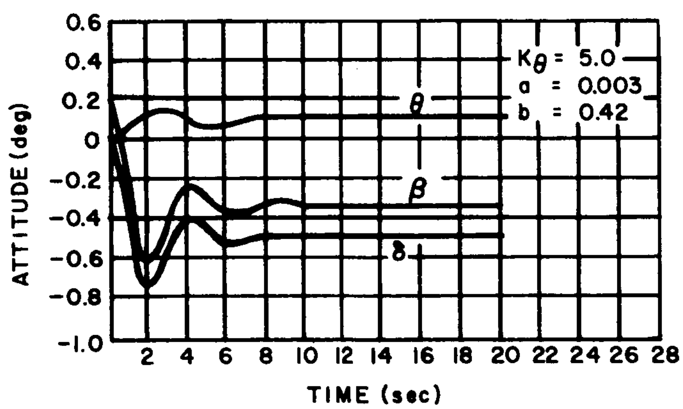


Figure 5.2-16. LEMDE Autopilot Response-Prior to Retro with Capsule On

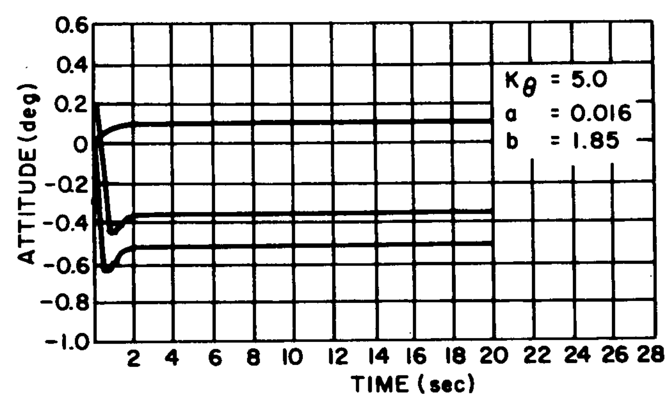


Figure 5.2-17. LEMDE Autopilot Response-After Retro with Capsule Off

5.2.3.2 Velocity Magnitude Control. As defined in VC220FD101 and summarized in Section 3.2 of VC234FD105, Volume A, the autopilot must control the velocity magnitude to within 1.8% (3 σ) or 0.052 meters/sec (3 σ), whichever is larger. To attain this accuracy, a roll axis accelerometer and integrator are used to control engine on-time. The integration is performed in two steps.

a. An analog integrator generates a pulse and resets itself each time an increment of approximately 0.01 ft/sec is attained

b. The Controller and Sequencer (C&S) digitally sums these pulses and compares the sum to a preset number.

When the desired velocity is attained, the C&S commands engine shut-down. A timer circuit will be used as a back-up to the accelerometer and integrator to proteot against catastrophic failures. Sources of error in the accelerometer technique of engine shut down are as follows:

a. Engine uncertainties

   1. Uncertainties in the delay between the generation of a shut-down command and actual valve closure.

   2. Engine tail-off uncertainty, i.e., deviation from the nominal thrust vector decay profile.

   These uncertainties, expressed as impulse, are as follows:

Configuration	Impulse Uncertainty (lb-sec)
Transtage Main Engines	780
Small Engines	19
LEMDE High Thrust	512
LEMDE Low Thrust	51

b. Accelerometer - integrator uncertainties (significant contributors only):

   1. Accelerometer scale factor error (9×10^{-5}) (g) (t)
   2. Integrator error (3×10^{-5}) (g) (t)
   3. Integrator bias (0.003 Δ V)
   4. Integration granularity (0.0009) meters/sec)

Where:

g is the acceleration level in g
t is the engine on-time

Errors in termination of the thrust were root-sum-squared with the accelerometer-integrator errors for seven representative maneuvers. The results are presented in Table 5.2-4. For purposes of comparison, the corresponding results for the preferred design are also included. In generating this table, it was assumed that only the retromaneuver used the large engines. All correction maneuvers were made with the small engines (low thrust for LEMDE). The last row shows the accuracy obtained if the transtage engines were used for all maneuvers. It can be seen that the required accuracy cannot be attained for velocity changes of 30 meters/sec or less. The first column of Table 5.2-3 shows the minimum velocity increment which can be performed with an accuracy of 0.9%. This error was based on earlier guidance studies. It should be noted that the permissible error is 1.8% so that somewhat smaller velocity increments could be performed with sufficient accuracy.

In summary, the transtage engines in conjunction with low thrust engines, and the LEMDE throttlable to its minimum thrust level both can meet the desired velocity magnitude control accuracy.

TABLE 5.2-4. PERCENT AND MAGNITUDE OF VELOCITY ERROR (3σ) IN PERFORMANCE OF TYPICAL MANEUVERS FOR CONSIDERED ENGINE CONFIGURATIONS

Propulsion System	Minimum ΔV For 0.9% Error 17,500 lb (meters/sec)	3σ% Error For 2 meters/sec 20,500 lb Vehicle	3σ% Error For 30 meters/sec 20,500 lb Vehicle	3σ% Error For 150 meters/sec 20,500 lb Vehicle	3σ Error For 2200 meters/sec	Orbit Trim Minimum ΔV For 0.9% Error 3200 Vehicle (meters/sec)	3σ% Error For 0.2 meters/sec 17,500 lb Vehicle
Preferred Design Solid 30,000 lb Liquid 400 lb	④⑤ 1.26	① 0.015 meters/sec or 0.75% ②	① 0.171 meters/sec or 0.5%	① 0.84 meters/sec or 0.56%	1.07% Total Impulse	6.62	0.0113 meters/sec ④ or 5.65%
Transtage 16,000 lb Liquid 400 lb	④⑤ 1.26	① 0.015 meters/sec or 0.75% ②	0.171 meters/sec or 0.5% ①	① 0.84 meters/sec or 0.56%	6.66 meters/sec or 0.302% ③	6.62	④ 0.0113 meters/sec or 5.65%
LEM 10,500 lb or 1050 lb	3.16 ⑤	0.0253 meters/sec or 1.27% ④	0.108 meters/sec or 0.36% ③	0.59 meters/sec or 0.394% ③	6.63 meters/sec or 0.302% ③	17.4	0.0284 meters/sec ④ or 14.2%
Transtage 16,000 lb	④ 48.8	0.375 meters/sec or 18.8% ④	③ ④ 0.384 meters/sec or 1.8%	① ② ③ ④ 0.906 meters/sec or 0.605%	③ 6.66 meters/sec or 0.302%	266	

PRINCIPAL CONTRIBUTORS TO ERRORS

① Accelerometer Scale Factor Error
② Integrator Error
③ Integration Bias
④ Engine Stop Errors
⑤ Integration Granularity

NOTE: Circled numbers in the chart indicate the dominant source of error for that calculation (cross referenced to the adjacent list)

5.2.4 Alternatives to the Transtage and LEMDE Autopilot Designs

5.2.4.1 Alternatives Common to the Solid Engine Design. Many alternatives considered for the preferred solid engine design are also applicable to the autopilot designs for transtage and LEMDE. Reference is made to Section 3.4 of VC234FD105, Volume A with the following comments.

a. Concerning alternatives indicated in Section 3.4.1 of VC234FD105 the same positive feedback circuit about the actuator may be considered. Although uncertainty in cg offset angles for transtage and LEMDE are slightly larger due to the liquid propellant, they are probably not large enough to justify the additional feedback.

b. Concerning alternatives indicated in Section 3.4.4 of VC234FD105, pitch and yaw engine biasing due to cg offset is somewhat larger, and may necessitate implementation of this alternative.

5.2.4.2 "Bang-Bang" Operation With Low (Four 25-Pound Thrusters) Fixed Engines. This system requires operation of the big engines for all maneuvers whose impulse exceeds the minimum impulse capability of the large engine (4200 lb-sec). Errors in operation of the large engines are taken out by the vernier operation of the low thrust system either during large engine firing or after large engine shut-down. Thrust vector control for the low thrust system requires no vanes or gimbals. Disadvantages are that the main engine will be less reliable because of the larger number of restarts. Also, a separate roll control system must be designed.

APPENDIX A

SPACECRAFT CHARACTERISTICS

This appendix describes representative vehicle characteristics for transtage and LEMDE during critical phases of the VOYAGER Mission.

In Appendix B, a vehicle transfer function is developed:

$$\theta = - \frac{as^2 (\delta + \delta_T) + b(\delta + \delta_T + \delta_{cg})}{s^2}$$

Where:

θ = vehicle roll axis angle with respect to an inertial reference

δ = engine gimbal angle

δ_T = misalignment of gimbal to roll axis

b = vehicle gain = $\frac{F \ell_1}{I}$

F = engine thrust level

ℓ_1 = distance from thrust application point to vehicle cg

$a = \frac{m_e \left[r_{ge}^2 + \ell_2 (\ell_1 + \ell_2) \right]}{I}$

r_{ge} = radius of gyration of the engine

m_e = engine mass

ℓ_2 = distance from engine cg to thrust application point

I = pitch or yaw moment of inertia

The term "a" has value only if gimballed engines are used; jet vanes have so little mass their inertial coupling effect is negligible.

Table A-1 summarizes this information, using the best estimate of cg location and moments of inertia. The "a" and "b" values are approximate and can be used with equal accuracy about the pitch and yaw axes.

TABLE A-1. SPACECRAFT CHARACTERISTICS

Engine Configuration	ℓ_1 (in.)	ℓ_2 (in.)	I (pitch) (slug-ft^2)	Large Engine		Small Engine	
				a	b	a	b
LEMDE							
a) Before retro, capsule on	89	1	17,000	0.003	4.2	0.003	0.42
b) Before retro, capsule off	64	1	4,000	0.013	14.0	0.013	1.4
c) After retro, capsule on	116	1	13,000	0.004	7.8	0.004	0.78
d) After retro, capsule off	63	1	3,000	0.016	18.5	0.016	1.85
Transtage (Unmodified)							
a) Before retro, capsule on	82	1	29,000	0.0026	3.8	$1.5 \cdot 10^{-5}$	0.095
b) Before retro, capsule off	53	1	11,000	0.0066	6.4	$2.6 \cdot 10^{-5}$	0.16
c) After retro, capsule on	140	1	16,000	0.0050	11.7	$4.6 \cdot 10^{-5}$	0.29
d) After retro, capsule off	86	1	7,000	0.0101	16.4	$5.5 \cdot 10^{-5}$	0.41
Transtage (Modified)							
a) Before retro, capsule on	65	1	19,000	0.0039	4.55	$1.9 \cdot 10^{-5}$	0.114
b) Before retro, capsule off	38	1	5,000	0.014	10.0	$4.5 \cdot 10^{-5}$	0.25
c) After retro, capsule on	95	1	14,000	0.0055	9.0	$3.7 \cdot 10^{-5}$	0.225
d) After retro, capsule off	42	1	3,650	0.02	15.0	$6.6 \cdot 10^{-5}$	0.375